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Volume III
Guideline Analyses

Prepared by ADVANCED VEHICLE SYSTEMS DIRECTORATE
Systems Planning Division

5 October 1973

Prepared for OFFICE OF MANNED SPACE FLIGHT
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Washington, D. C.

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2

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PAYLOAD DESIGN REQUIREMENTS ANALYSIS (STUDY 2.2)
FINAL REPORT
Volume III: Guideline Analyses

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FOREWORD

This report documents The Aerospace Corporation effort on Study 2.2, Payload Design Requirements Analysis, performed under NASA Contract No. NASW-2472 during Fiscal Year 1973. The Aerospace study was monitored by Dr. R. W. Johnson, NASA Headquarters; J. O. Ballance, Marshall Space Flight Center; and R. A. Berglund, Johnson Space Center, and their efforts as a team in providing technical direction throughout the duration of the study are greatly appreciated.

This volume is one of three volumes representing the final report of Study 2.2. The three volumes are:

Volume I	Executive Summary
Volume II	Payload Design Guidelines
Volume III	Guideline Analyses

Volume I summarizes the overall report in brief form and includes the relationship of this study to other NASA efforts, significant results, study limitations, suggested research, and recommended additional effort.

Volume II provides the design guidelines in concise format with sufficient information to permit tradeoff results. It also includes the application of the guidelines to an example satellite as a demonstration of their usefulness.

In Volume III, all of the analyses that were performed are documented to provide traceability. These analyses include analytical technique, design analyses of the Large Space Telescope and the Shuttle-Launched Defense Support Program (SLDSP) payloads, common hardware, and Sortie payload operations. (Figures showing conceptual design of the SLDSP were intentionally left out of Volume III for security reasons, but they are available from the Study Director upon establishment of need to know.) The subsystem analyses are presented in the appendixes of Volume III.

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CONTENTS

1. INTRODUCTION 1
 A. General 1
 B. Background 2
2. ANALYTICAL TECHNIQUE 5
 A. General 5
 B. Spacecraft Synthesis Program 7
 C. SAMS System Optimization 1[illegible]
 D. Launch and Payload Service Rates. 29
 E. Payload Program Cost Estimating 31
3. DESIGN ANALYSIS 43
 A. Introduction 43
 B. Large Space Telescope Design Analysis 44
 C. Shuttle-Launched Defense Support Program Design Analysis 91
4. COMMON HARDWARE 131
 A. Introduction 131
 B. Attitude Control Hardware 134
 C. Solar Array 156
 D. Conclusions 171
 E. Recommendations. 173
5. SORTIE PAYLOAD OPERATIONS 175
 A. Introduction 175
 B. Analysis of a Joint ASF/SSO Payload. 176
 C. The Modified Atmospheric Sciences Facility 183
 D. Integration at a Sortie Center 187
 E. Orbital Operations 187
 F. Ground Operations 194
 G. Sortie Payload Capability. 203
REFERENCES. 205

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CONTENTS (Continued)

APPENDIXES

A. STRUCTURAL DESIGN A-1

B. LOW-COST SPACECRAFT STRUCTURAL DESIGN CRITERIA GUIDELINES FOR SHUTTLE-LAUNCHED PAYLOADS B-1

C. STRUCTURE COMPLEXITY RATING C-1

D. FACTOR OF SAFETY FOR PRESSURE VESSELS D-1

E. THERMAL CONTROL E-1

F. REACTION CONTROL F-1

G. ELECTRICAL POWER G-1

H. ELECTRICAL DISTRIBUTION H-1

I. BACKUP SYSTEM FOR RETRIEVAL I-1

J. GROUND OPERATIONS TIME J-1

K. LAUNCH SITE TESTING K-1

L. DESIGN L-1

M. RELIABILITY M-1

N. FMEA PROCEDURE N-1

TABLES

2-1. Basic Weight Equations 11

2-2. Shuttle/Tug Launch and Payload Rate Equations 32

2-3. Satellite Cost Estimate - Basic Input Information 35

3-1. LST Stabilization and Control Subsystem Failure Rate, Parts List, and Weight 54

3-2. LST Electrical Power Subsystem Failure Rate, Parts List, and Weight 55

3-3. LST Communication and Data Management Subsystem Failure Rate, Parts List, and Weight 56

3-4. LST Scientific Instrument Package Subsystem Failure Rate, Parts List, and Weight. 57

3-5. LST Optical Telescope Assembly Subsystem Failure Rate, Parts List, and Weight 58

3-6. LST Structural Weights for On-Orbit Maintainable Payload 73

3-7. Shuttle-Launched LST Weight, Expendable Payload. 74

3-8. Shuttle-Launched LST Weight, Ground-Refurbishable Payload. 75

3-9. Shuttle-Launched LST Weight, On-Orbit Maintainable Payload 76

3-10. LST Space-Replaceable Module Characteristics for a 24-Module Concept 77

3-11. SLDSP Parts Quantity List 103

3-12. Required Volume Per Module for Subsystem Packaging . . 109

3-13 SSPRO Weight Prediction for the SLDSP Payload. 118

TABLES (Continued)

3-14. Weibull Functions for SLDSP Modules 121

4-1. Cost Savings from Common Hardware 132

4-2. Type 3 Vehicles, Class I Missions 138

4-3. Group 1 140

4-4. Group 2 141

4-5. Group 3 142

4-6. Group 4 143

4-7. Group 5 144

4-8. Group 6 145

4-9. Group 7 146

4-10. Group 8 147

4-11. Group 9 148

4-12. Group 10 149

4-13. Group 11 150

4-14. Group 12 151

4-15. Group 13 152

4-16. Group 14 153

4-17. Applicable Current Hardware 157

4-18. Common Hardware Assemblies for Stabilization Systems . 159

4-19. Sizing Results for 1 KW Solar Array 161

TABLES (Continued)

4-20.	General Characteristics of Custom and Standard Solar Array Modules	165
4-21.	Solar Array Planning Model	167
4-22.	Solar Panel Cost Breakdown	171
4-23.	Estimate of Total Solar Array Cost, 1979-1990	172
5-1.	ASF Facility - Baseline of Martin Story	177
5-2.	Candidate List for Instruments for Inclusion on a Solar Shuttle Observatory	178
5-3.	Solar Shuttle Observatory Instrument Field of View	180
5-4.	Comparison of Requirements	181
5-5.	Added Instrumentation for the Modified ASF	185
5-6.	Integration Data at a Sortie Center (Probably a Launch Site)	188
5-7.	Orbital Operations	190
5-8.	Experiment Equipment Main Assemblies and Complementary Equipment	197
5-9.	Evaluation of Operational Compatibility	204

FIGURES

2-1. Computational Flow Diagram 7

2-2. Spacecraft Synthesis Computer Program Flow Diagram . . . 9

2-3. Structure Weight Correlation 15

2-4. Structure Modularity Factor 17

2-5. Simplified Illustration of Shuttle Service Simulation 21

3-1. MSFC Large Space Telescope (LST) 45

3-2. LST Scientific Instrument Package (SIP) Reliability Model (No Redundancy) 46

3-3. LST Optical Telescope Assembly (OTA) Reliability Model (No Redundancy) 47

3-4. LST Attitude Control Subsystem Reliability Model. 48

3-5. LST Electrical Power Reliability Model 49

3-6. LST Communication and Data Handling Reliability Model 50

3-7. LST Weight Growth with MMD 52

3-8. Modularized LST, Concept No. 1 59

3-9. Modularized LST, Concept No. 2 60

3-10. Modularized LST, Concept No. 3 61

3-11. Modularized LST, Concept No. 4 62

3-12. Modularized LST, Concept No. 5 63

3-13. Swing Translation to Replace Module for On-Orbit Maintenance Using Orbiter RMS 65

3-14. Space-Serviceable LST Spacecraft Concept. 69

FIGURES (Continued)

3-15. Modularized LST, Concept No. 5 80

3-16. Reliability Characteristics for LST Space-Replaceable Module No. 1 81

3-17. LST System Reliability Based on Weibull Function 82

3-18. Expected Launch Rate for LST 83

3-19. Effect of Mean Mission Duration on LST Unit and Program Cost 85

3-20. Effect of Program Duration on Total Program Cost for LST 86

3-21. Effect of Repair Cost Factor on Total Program Cost for LST 88

3-22. Effect of Trip Charge on Program Cost for LST 89

3-23. Optimum Number of Modules Replaced for LST with 22 Space-Replaceable Modules 90

3-24. SLDSP Structures and Propulsion Reliability Model 9[illegible]

3-25. SLDSP Electrical Power Reliability Model 9

3-26. SLDSP Communication Link 1 Reliability Model 95

3-27. SLDSP Communication Link 2 Reliability Model 96

3-28. SLDSP Communication Link 3 Reliability Model 97

3-29. SLDSP Communication Data Link Reliability Model 98

3-30. SLDSP Communication Data Transmission Reliability Model 99

3-31. SLDSP Communication Command Transmitter and Receiver Reliability Model 100

3-32. SLDSP Attitude Control Reliability Model 101

FIGURES (Continued)

3-33. SLDSP Weight Growth with MMD 102

3-34. Space-Serviceable Spacecraft (Eight-Module Configuration) 107

3-35. SLDSP Spacecraft (13-Module Configuration) 111

3-36. SLDSP Spacecraft Two-Module Configuration 115

3-37. SLDSP Weight Factors Versus MMD 120

3-38. Expected Launch Rate for a Two-Module SLDSP 122

3-39. Expected Launch Rate for an Eight-Module SLDSP 123

3-40. Expected Launch Rate for a 13-Module SLDSP 124

3-41. Effect of Mean Mission Duration on SLDSP Unit and Program Cost. 125

3-42. Effect of Modules Replaced on Total Program Cost for SLDSP with Two and Eight Space-Replaceable Modules . . . 127

3-43. Effect of Modules Replaced on Total Program Cost for SLDSP with 13 Space-Replaceable Modules. 128

3-44. Effect of Number of Modules in SLDSP on Total Program Cost. 129

4-1. Classification of Attitude Control Subsystems 137

4-2. Solar Cell I-V Curves (10 Ω-cm, 2x2 cm) 162

4-3. 1 Ω-CM Solar Cell I-V Curves 163

4-4. Solar Array Recurring Cost 168

4-5. Silicon Solar Cells Learning Curve 170

5-1. A Slave Satellite Concept 186

FIGURES (Continued)

5-2. Time Lines Measured from Experiment Start (Not Launch Time) 193

5-3. Payload Operations Flow Cycle 195

5-4. Modified ASF Orbital Configuration 196

5-5. Instrument Integration Time Lines 199

5-6. Control Consoles Integration Time Lines 200

5-7. Experiment Equipment Removal 202

1. INTRODUCTION

A. GENERAL

Prior studies have indicated that the Space Transportation System (Shuttle plus Tug) should produce a large reduction in overall payload costs as a result of improved payload design and operational techniques. This study was conducted to provide data and insight into the methods of achieving these economic gains.

The study treats only payloads that will be launched on the Shuttle/Tug/Sortie Lab combinations. These payloads are of four types:

Expendable
Ground Refurbishable
On-Orbit Maintainable
Sortie

The expendable payloads addressed in this study are designed specifically for the Shuttle/Tug and not for expendable launch vehicles. Economic comparisons are made only between these four types of Shuttle payloads, and not between these payloads and current expendable launch vehicle payloads.

The FY 1972 study that preceded this study identified a series of design guidelines that were documented in a Reusable Payload Specification report (Ref. 1). The FY 1973 study attempted to quantify several of these guidelines that were identified to be cost effective. Additional system guidelines were also developed in this study by analyzing two payloads parametrically and demonstrating the results on an example satellite. In addition to analyzing the selected guidelines, emphasis was placed on providing economic tradeoff data and identifying payload parameters influencing the low cost approaches. Thus, in no way should this study be considered to represent a point design effort. The economic analyses reported in these volumes should be viewed as providing trend data rather than absolute cost data.

This volume documents the analyses that were performed in developing the design guidelines prsented in Volume II, Payload Design Guidelines. The analytical techniques and design analyses that are presented in this volume provide traceability in the approach, data generation, and data flow for the design guidelines on low-cost-type payloads. The analyses on the other guidelines such as reliability, modularity, ground operations, and subsystems are presented in the appendixes since they were performed independently.

The common hardware and Sortie payload operations tasks were special emphasis tasks in this study. The common hardware concept has a potential for large cost reductions and was therefore examined in detail to determine its technical feasibility. This analysis was favorable for Shuttle payloads and resulted in a design guideline. The Sortie payload operations task was conducted to investigate the integration of two scientific disciplines into a multiexperiment Sortie payload. When the two disciplines selected for integration were found to be incompatible, the mission capability was expanded in one of the disciplines to update the Sortie mission by incorporating additional experiments.

B. BACKGROUND

During Fiscal Year 1972, a payload analysis for Space Shuttle applications was conducted by The Aerospace Corporation to identify design guidelines for reusable payloads (Ref. 1). The FY 1972 payload design guidelines resulted principally from an overall review by specialists experienced in the design of satellites of Shuttle payload studies performed to that time.

In continuing this work during FY 1973, The Aerospace Corporation was able to use more definitive data on the Shuttle, Tug, Sortie Lab, and payloads from NASA and DOD studies to develop additional guidelines and to quantify the FY 1972 guidelines. Since only a portion of the FY 1972 guidelines could be examined in depth within the scope of the FY 1973 effort,

it was decided to analyze only one guideline pertaining to each subsystem along with a number of system guidelines. Each subsystem specialist analyzed his guideline from the viewpoint of technical feasibility or cost. The system guideline analysis on low-cost-type payloads was performed in terms of cost. Other guidelines were primarily analyzed for technical feasibility with some supporting qualitative cost information.

It was the intent to analyze as many of the guidelines as possible to see how they would apply to a Shuttle payload, what tradeoffs would arise, and what guidelines would prove practical for a serviceable payload.

2. ANALYTICAL TECHNIQUE

A. GENERAL

This effort was performed to develop analysis techniques for determining total payload program cost as a function of payload parameters. These cost relationships were found by developing a mathematical model to perform the parametric tradeoffs on the computer. This task was also used to identify specific tradeoffs that should be carried out because of their sensitivity to program costs. To relate the parametric analysis to actual cases, two satellites were selected to represent a low altitude mission and a high altitude mission. This section describes the mathematical model, computational flow, and parameters involved in the trade studies. The cost trend data and the sensitivity of the parameters are presented in Section 3.

The computational flow of the mathematical model is shown in Figure 2-1. The reliability model and module simulation program are described in the SAMS Systems Optimization Program later in this section as are the Spacecraft Synthesis Program and the Payload Cost Program. Because of the large quantity of data that is produced, a data analysis and summarization program was developed to search the stored cost data in the Payload Cost Program to identify the lowest cost concepts for a given set of conditions.

Also shown in the computational flow diagrams are the parameters that were varied in each of the computer programs. These parameters were systematically varied to determine their cost sensitivities for a given mission description. The remote terminal computer facility was used to provide computational flexibility during the development of SAMS. The payload cost program was converted from CDC/FORTRAN to the remote terminal/advanced programming language (APL) to also provide development flexibility. Output of each program was printed, and the input into the next computer program

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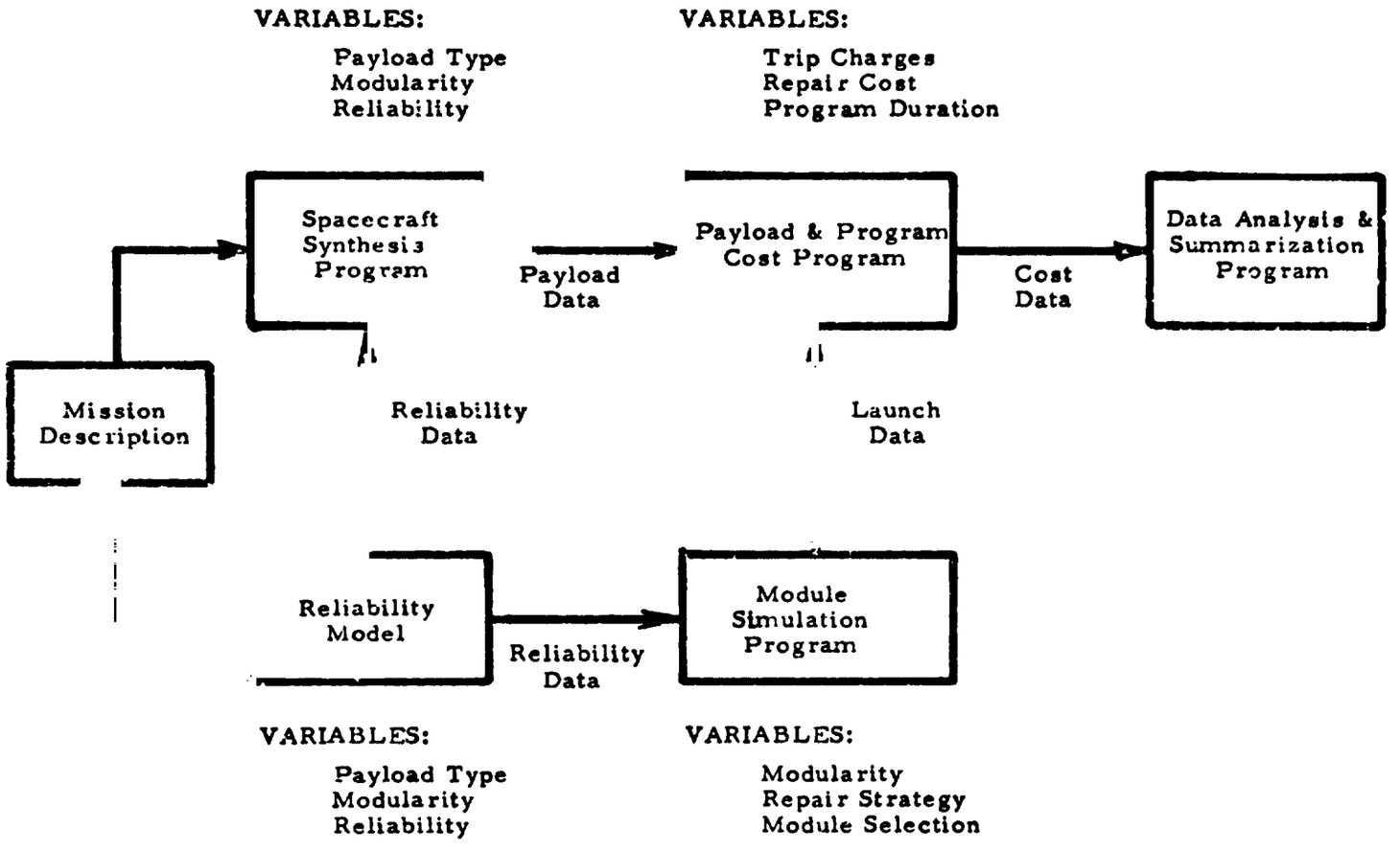


Figure 2-1. Computational Flow Diagram

was automatic except for those programs where there was not sufficient workspace in the remote terminal to store the data. In those cases where the data exceeded the workspace, the inputting was performed manually.

Within each program in the mathematical model, analysis was required to develop appropriate factors or coefficients for the computation to process the data. In the case of the reliability model, data on the reliability diagrams, failure rates, type of redundancy, and parts weights were required to develop the payload subsystem redundancy and reliabilities. The Spacecraft Synthesis Program required design factors to convert from expendable to ground-refurbishable and on-orbit maintainable payloads. The Payload Cost Program required redundancy cost factors to account for changes in mean mission duration (MMD) from the baseline payload. The development of these factors and equations are described in this section.

B. SPACECRAFT SYNTHESIS PROGRAM

1. INTRODUCTION

This computer program was developed to provide the necessary payload data to calculate payload costs which can be related to certain parameters of Shuttle payloads. The parameters of interest to the study include the effect of ground refurbishability, variation of MMD, and the number of modules in spacecraft designed for on-orbit maintenance. Some preliminary analyses were also conducted to determine factors for Sortie missions. The Spacecraft Synthesis Program also had the capability to vary basic parameters such as electrical power, pointing accuracy, propellant type, velocity increment, etc. Where necessary, the spacecraft shape could be modified to be compatible with the Orbiter cargo bay. The spacecraft data resulting from the variation of these parameters would be used as primary inputs to cost analysis routines.

The Spacecraft Synthesis Program was developed jointly with support from Study 2.4 and is thus more extensive than if it had been developed for either study alone.

The Spacecraft Synthesis Program can be used to develop subsystem data for most satellites. In this study, these satellites include the Shuttle-Launched Defense Support Program (SLDSP) and the NASA Large Space Telescope (LST) and Earth Observatory Satellite (EOS) program. This section of the report describes the development and operation of the Spacecraft Synthesis Program and the equations.

2. SYNTHESIS PROGRAM DESCRIPTION

The Spacecraft Synthesis Program has been prepared in both Fortran IV and APL format and can be used on both the CDC 7600 and the IBM 370 at The Aerospace Corporation. English rather than metric units are used to facilitate interface with other computer routines. Where convenient, actual values for parameters of lesser influence have been incorporated as fixed values. Where the influence of the variable is a complex function, the equation for the influence is included in the program and is automatically treated. For example, the modularity variable is defined by an exponential equation. In using the program, the operator need only input the number of modules to be studied. These automatic features reduce errors in the program operation and expedite the computations.

The computer develops subsystem weight and other payload data as functions of the input parameter values. Equipment subsystems are computed first. The structure and thermal protection subsystems, which are partially dependent on the equipment subsystem data, are produced next by iteration subroutines. The final step is the development of the payload adapter weight and length.

A simplified flow diagram for the Synthesis Computer Program is depicted in Figure 2-2. The equations developed for the program are delineated and described in the ensuing paragraphs. The completed program is, of course, tested against existing satellites periodically.

Two types of subsystem weight estimating equations were developed for use in the synthesis program. The first type includes basic equations

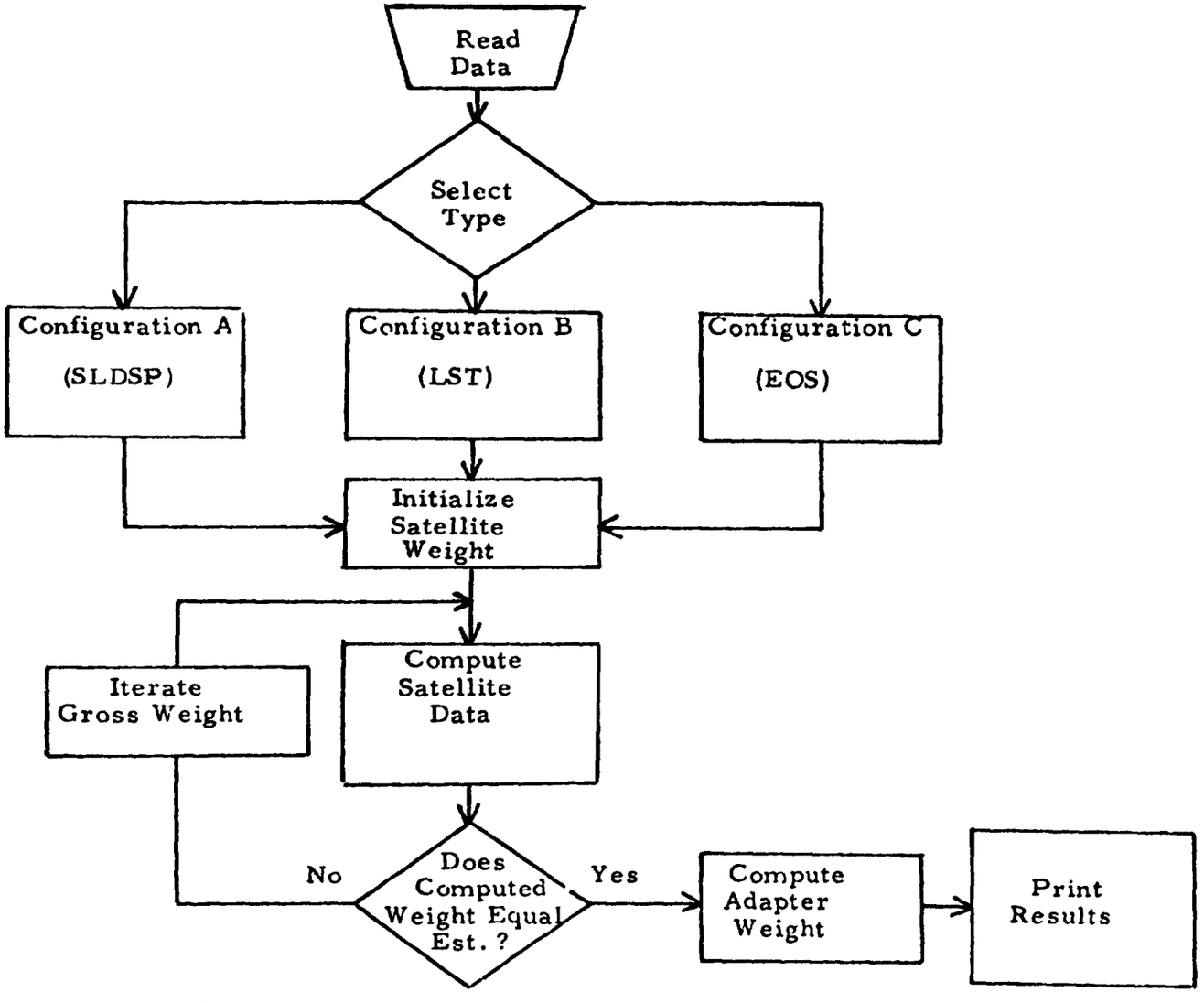


Figure 2-2. Spacecraft Synthesis Computer Program Flow Diagram

for estimating the weights of subsystems of current expendable satellites. Actual satellite data were collected from NASA and DOD programs to permit generation of these basic equations. The second type of equations offers, in essence, factors for modifying the subsystem weights for Shuttle payloads.

The basic subsystem weight estimating equations were developed by establishing an analytical model and correlating data from existing satellites with the model to produce the equations. The data were correlated using a regression analysis computer routine. Parameters having a low, or constant, influence were incorporated as fixed values for simplication. The basic weight equations for the subsystems are delineated in Table 2-1. The symbols are described as they first appear. The correlation of actual satellite data with the developed equation is shown in Figure 2-3 for the structure subsystem. These equations are continually being refined and expanded as data become available. For this study, the equations that were available on 28 February 1973 were used and were not updated.

The use of the basic equations will permit computation of data for satellites of the expendable type that are now being fabricated for launch on expendable vehicles. To modify these designs for Shuttle applications, Shuttle application factors, in equation or coefficient form, were developed for the subsystem equations within the program.

Ground-refurbishable payloads are designed to permit on-orbit retrieval and repair or updating on the ground. The synthesis program provides for these features by adding a docking structure to the expendable satellite 45.4 kg (100 lb), adding a targeting component 4.5 kg (10 lb), and increasing the structure by a factor of 1.10 to provide increased access to interior components.

Mean Mission Duration (MMD) and design life of payloads were varied for Shuttle applications. To incorporate this feature, studies were conducted which determined the increase in the number of components

Table 2-1. Basic Weight Equations

1. Structure

$$W_s = K_\rho \left((W_c)^{0.9} (L/_D)^{0.24} \right)^{1.096}$$

where:

K_ρ = Density Coefficient
= 0.219 for satellites with body-mounted solar cells (ENDO)
= 0.128 for satellites with extendable solar panels (EXO)

W_c = Weight of Satellite contents, pounds

$L/_D$ = Satellite length-to-diameter ratio

2. Thermal Control (Passive)

$$W_{tc} = 0.025\ W_{sc}$$

where:

W_{sc} = Satellite weight, pounds

3. Electrical - Batteries

$$W_{bat} = 0.167\ (P_{bat})\ (T_e)\ (1 + R)\ (0.98)^{(IOC-1960)}$$

where:

P_{bat} = Battery power required during eclipse, watts

T_e = Time in eclipse, hours per orbit

R = Redundancy factor (e.g. R = 0.5 = 50%)

IOC = Year of initial operational capability

Table 2-1. Basic Weight Equations (Cont'd)

4. Solar Arrays (Synchronous Orbits Only)

Body mounted:

$$W_{s/a} = \left[\frac{0.264\ P_{s/a}}{0.86 - 0.0314\ Life}\right]\left[\frac{0.38}{PF} + 0.35\right]\left[0.99^{(IOC-1960)}\right]$$

Oriented Paddles - Rigid Substrate:

$$W_{s/a} = \left[\frac{0.120\ P_{s/a}}{0.86 - 0.0314\ Life}\right]\left[\frac{1}{PF} + 0.35\right]\left[0.99^{(IOC-1960)}\right]$$

Oriented Panels - Flexible Substrate:

$$W_{s/a} = \left[\frac{0.109\ P_{s/a}}{0.86 - 0.314\ Life}\right]\left[\frac{0.4}{PF} + 0.35\right]\left[0.99^{(IOC-1970)}\right]$$

where:

$P_{s/a}$ = Total solar array power requirement, watts

Life = Design life of spacecraft, years

PF = Ratio of solar cell-to-substrate areas

5. Electrical Harness

$$W_h = 0.0149\ (W_{eq})^{1.357}(V_{sc})^{0.073}$$

where:

W_{eq} = Weight of power consuming equipment, pounds

V_{sc} = Volume of spacecraft, cubic feet

Table 2-1. Basic Weight Equations (Cont'd)

6. **Electrical - Power Conditioning**

$$W_{pc} = 0.386 (P_{s/a})^{0.73}$$

7. **Guidance, Navigation and Stabilization**

Three axis control:

$$W_{gns} = 0.648 (W_{sc})^{0.67} (PA)^{-0.14}$$

Spin stabilization:

$$W_{gns} = 0.558 (W_{sc})^{0.60} (PA)^{-0.20}$$

where

W_{sc} = Weight of spacecraft on orbit, pounds

PA = Pointing accuracy, degrees

8. **Reaction Control Propellants - N_2H_4**

$$W_p = 0.0273 W_{sc} (Life)^{0.814}$$

9. **Reaction Control Subsystem Hardware - N_2H_4**

$$W_{rcs} = 0.619 (W_p)^{0.844}$$

Table 2-1. Basic Weight Equations (Cont'd)

10. Communication, Data Processing, Instrumentation

$$W_{cdpi} = 50 + 5(H)^{0.1} (N_{dl}-1) + 15 N_{tr} + D_p + ENC$$

where:

- H = Orbit altitude, nautical miles
- N_{dl} = Number of down links
- N_{tr} = Number of tape recorders
- D_p = Data processing equipment weight, pounds
- ENC = Encryption equipment weight, pounds

11. Adapter Weight

$$t^* = \left\{ \frac{W_g \left(3 + \frac{4\ell_g}{D}\right)}{2\pi E + 9 \left(\frac{2t}{D}\right)^{0.6} + 0.16 \left(\frac{D}{2\ell_a}\right)^{1.3} \left(\frac{2t}{D}\right)^{0.3}} \right\}^{0.5}$$

$$W_{adapt} = K_{ad} \pi D \ell_a t \rho$$

- W_g = Load on adapter, pounds
- D = Adapter diameter (average), feet
- ℓ_a = Adapter length, feet
- ℓ_g = Centroid of adapter load to centroid of adapter, feet
- t = Adapter shell thickness, feet
- E = Modulus of elasticity
- ρ = Material density, pounds per cubic foot
- K_{ad} = Ratio of realistic-to-theoretical weight; 1.5
- * = Automatically iterated within program

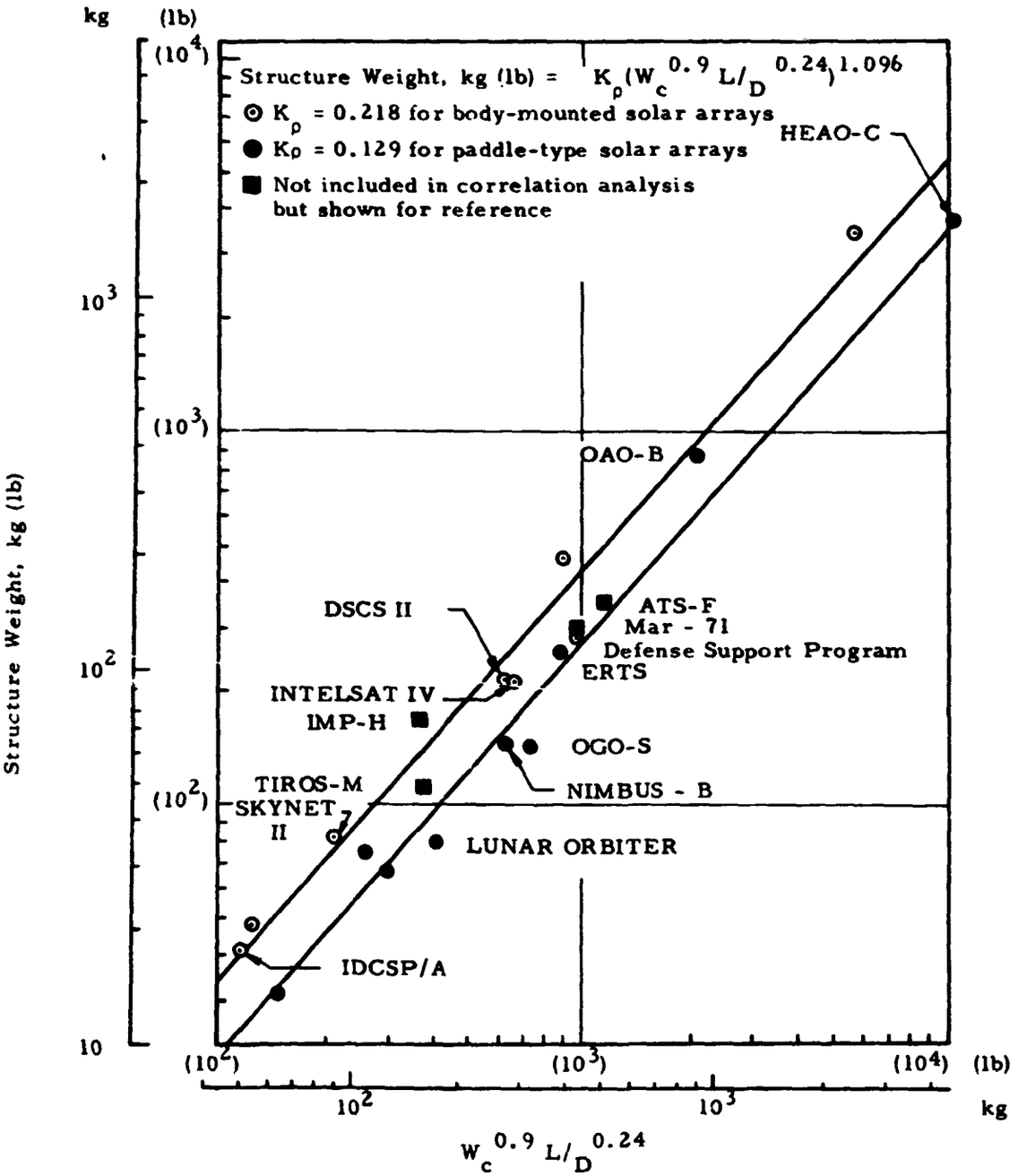


Figure 7-3. Structure Weight Correlation

required in a system as a function of the increase in reliability. Data from these studies, which are presented in Section 3, were converted to weight factors for use in the synthesis program. Propellants and associated tankage weights were also increased as design life increased. For this study, design life was set to equal 1.8 × MMD.

On-orbit maintenance of payloads was accomplished by incorporating replaceable modules in the payload design. This feature is accomplished within the synthesis program by applying factors to the appropriate subsystems. These factors were developed from design analysis conducted during the study as follows:

Subsystem	Factor
Structure	
with fewer than 8 modules	= 0.1143 N_m + 0.8857
with more than 8 modules	= 0.0875 N_m + 1.10
Electrical Dist. and Conditioning	= 19.7 N_m
Thermal Protection	= 1.10 W_{tc}

where:

N_m = Number of modules

W_{tc} = Thermal protection subsystem weight, pounds

A graph of the structural modularity factor is shown in Figure 2-4 which shows the correlation of data supplied in Reference 2 and the design studies performed at The Aerospace Corporation.

These weight equations are only applicable over a limited range. These limits are:

Weight < 1000 kg (2205 lb)
Electrical Power < 5000 watts
Design Life < 10 years
Pointing Accuracy + 1.7×10^{-4} radians (0.01 deg)

SEO = Synchronous Earth Orbiter
EOS = Earth Observatory Satellite
OAO = Orbiting Astronomical Observatory
DSCS = Defense Satellite Communications System
TAC = The Aerospace Corporation Study
(Modified Defense Support Program)

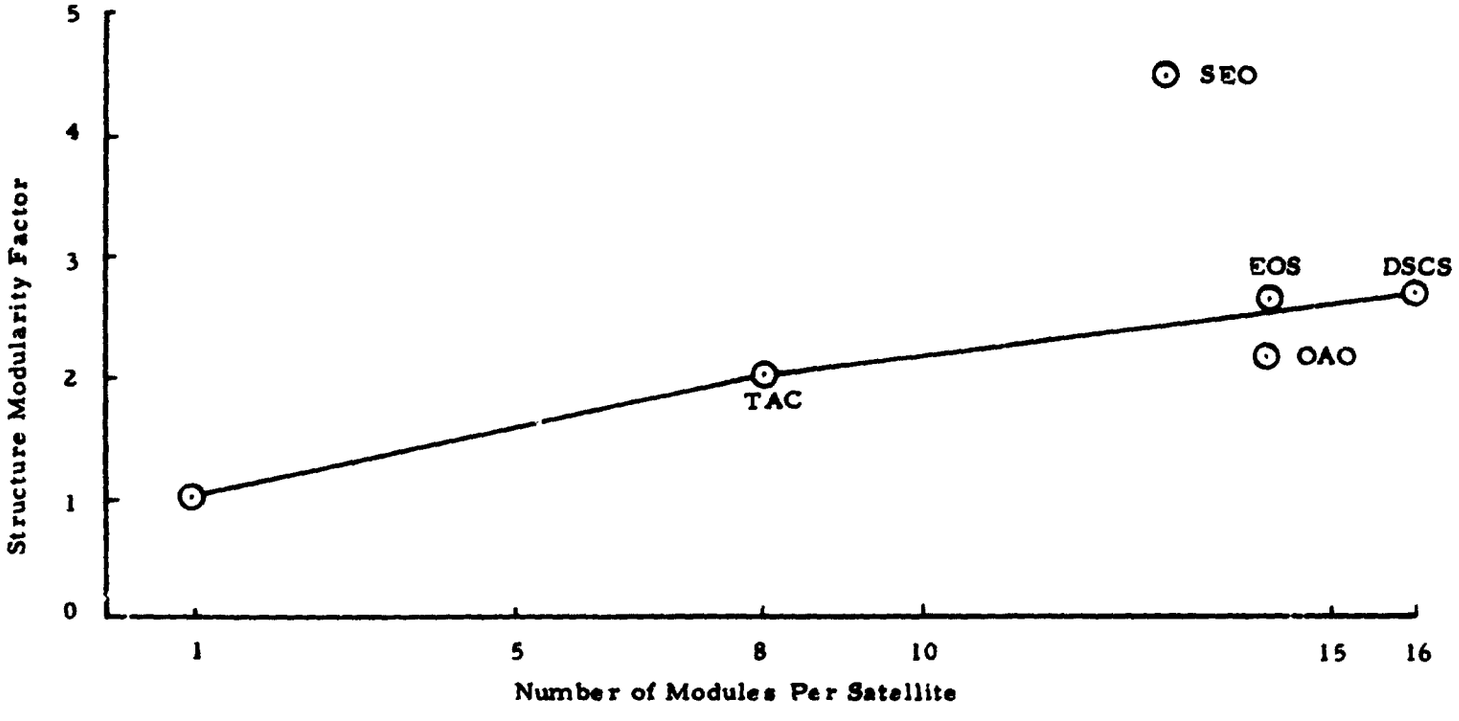


Figure 2-4. Structure Modularity Factor

3. SYNTHESIS PROGRAM OPERATION

The synthesis program prints data in tabular form. The first column lists the reference weights for a current-technology, expendable launch vehicle satellite. Weights are listed by subsystem. The second column shows the subsystem weights as the payload shape is modified to fit the Shuttle cargo bay. The third column delineates the subsystem weights of a Shuttle-launched payload with retrieval and ground-refurbishability features. The fourth column presents the subsystem weights of a Shuttle-launched payload with on-orbit maintenance capability. The data are based upon the number of modules specified. The MMD variation effects are included in these three columns. Adapter weights are also provided. A typical computer run would include weights for a large variety of parametric values.

C. SAMS SYSTEM OPTIMIZATION

1. INTRODUCTION

The prime objective of the Steady State Availability of a Modularized System (SAMS) optimization analysis is to determine the frequency of Shuttle launches required to establish the maintain the payload complement in orbit. It is desired also to determine module usage rates and to assess the payload operational availability. The technique of simulation was selected as opposed to the closed-form approach because of the complexities involved in the analysis. Complexities such as non-homogeneous and non-exponential modules, launch strategies based on warnings as well as on failures, and repairs or renewals based on various degrees of predictive information are all readily analyzed by the simulation technique but are extremely difficult in the closed-form approach.

The simulation is Monte Carlo in that repeated trials of Shuttle operations are simulated in order to obtain the ensemble averages or other properties of the simulation. Briefly, the payload module failure times are generated at random according to their individual reliability functions, and,

depending on the launch and renewal strategies used, the failed modules together with other possibly degraded modules are repaired by generating new failure times for these modules. The simulation also considers whether the satellite is the expendable type, retrievable type, or on-orbit maintainable type.

The major inputs to the SAMS simulation are the module reliability functions. Each module has associated with it a Weibull reliability function of the form:

$$R(t) = e^{-\left(\frac{t}{\alpha}\right)^{\beta}}, \quad 0 < t < T$$

where:

α is a scale parameter having units of time

β is a non-dimensional shape parameter

T is the module truncation time.

The module reliability functions are obtained by first optimizing the payload redundancy using the System Optimization (SYSOPT) computer program, then assigning the various components to specific modules, and, finally, computing the Weibull reliability function for each module. Also considered are warning functions which are modelled by Weibull functions. The warnings are used primarily in the selection of non-failed modules to be loaded on the Shuttle for on-orbit repair.

Three methods of generating the pair of warning and failure times can be simulated. These methods are described in detail in the following section on Program Inputs and Life Simulation Methods. The methods depend on the definition of a warning and its relation to a failure. The simulation can, however, be used without considering the concept of warning.

The method of selecting the order of modules to be loaded on the Shuttle is an important concept in the simulation. The replacement order assumes failures first, then warnings, and, finally, if additional loading is

possible, selection of the additional modules based on using one of several predictive modes. In the predictive mode, a priori tagged modules are selected first, but untagged modules are never selected. In the random predictive mode, all remaining modules have an equal chance of being selected. In the perfect-failure-prediction (PFP) mode, the selection is made in increasing order of generated future failure times. In the conditional life mode, the selection is made according to the median value of their remainng lives. There are also other modes possible, depending on how much telemetry information is available, but these tend to be a compromise between the PFP and the conditional life modes.

The simulation as developed is described in Reference 3 and provides an efficient basis for performing complex replensihment analysis (involving the design as well as operational parameters) of both the Shuttle/Tug and payloads. The following sections provide general information on the SAMS program.

2. GENERAL SIMULATION CONCEPT

The primary purpose of the simulation is to obtain the following output parameters.

a. Payload availability
b. Payload module usage
c. Shuttle launch rate

These output parameters are described later in this section.

A typical Shuttle service simulation is shown in Figure 2-5. Two service actions are illustrated in the figure: the first can be a delivery-retrieval operation, and the second an on-orbit repair operation. In this case, the on-orbit repair operation can include both the repair of a failed module and the repair of degraded modules.

The simulation involves the establishment and maintenance of a four-payload system on orbit. The simulation assumes that the payloads

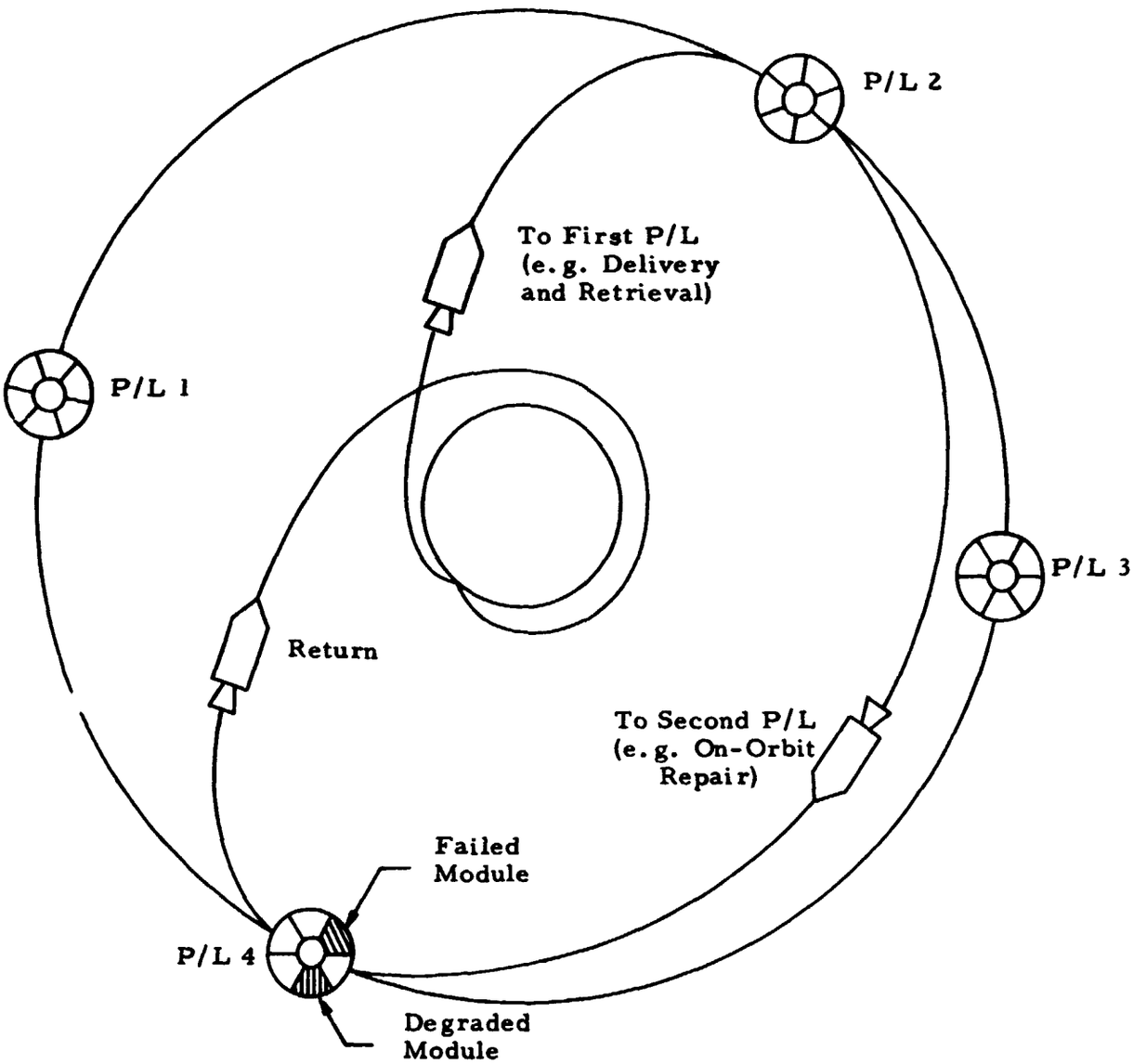


Figure 2-5. Simplified Illustration of Shuttle Service Simulation

are established in specified orbit positions at fixed launch times (the establishment mode) and that the maintenance mode begins after all the payloads have been placed on orbit. An establishment failure due to a launch failure or an early payload failure calls for servicing in the maintenance mode on a first failed-first served basis.

Each payload is assumed to be modularized with associated weight and reliability parameters. In addition to payload and module weights, SAMS considers the Tug performance parameters and the payload orbit parameters in order to arrive at "best" service loads for each launch. A "best" service load is the maximum total payload and module weight that the Tug can carry for each launch, consistent with the renewal strategy used.*

The "heart" of the simulation involves the module reliability functions which determine the life values of the modules. The module reliability functions are established using the SYSOPT computer program, and are discussed in the next paragraph. The method of simulating life values is described later in the section.

3. SYSOPT COMPUTER PROGRAM**

System Optimization (SYSOPT) is a Fortran computer program which takes any satellite reliability configuration and adds redundancies at the component (or black box level) to increase the MMD of the total system. The computer program systematically adds a redundant component, including expendables, one at a time for the greatest increase in MMD per unit weight. The candidate giving the greatest MMD/Δ weight ratio is selected for the additional redundancy. The computer program repeats the process of selecting redundancies until:

a. A maximum weight is reached.

b. The greatest MMD/Δ weight is less than 10^{-6} hours/kilogram.

c. 200 iterations have been completed.

* In this study, only a single payload service was assumed.

** This program was not developed in this study and is described in this report for completeness only.

At present, the types of redundancies in the computer program are:

are:

a. Active Redundancy

 All components are active, and only one operational component is necessary. Receivers are generally made active redundant.

b. Standby Redundancy

 Only one component is active, and the remainder are in a standby mode. Thus, the failure rates of the standby components are reduced to reflect the passive state. A sensing and switching device is necessary to detect failures of the primary mode. Transmitters are standby redundant.

c. Binomial Redundancy

 All n components remain active, and m components are required for nominal performance ($m \leq n$). Batteries in the power subsystem are usually redundant in this manner.

d. Normal Redundancy

 One component is active with failure described by normal density function. The failure rate of redundant components in the standby mode is modeled as zero in this study. Expendables are generally treated in this manner.

The information required to use the computer program is a list of components by subsystems which comprise the satellite system. For each component, the data needed are:

a. The components in the initial configuration

b. The maximum number of redundancies permissible

c. The type of redundancy and the minimum required in the case of the binomial redundancy

d. The duty cycle of the active component

e. The weight of each component

f. The added weight when the first redundancy is added

g. The active and standby failure rates (For the normal Redundancy model, these are the mean and standard deviation.)

The SYSOPT output identifies the subsystem and component and shows the quantity of components in the initial and final configuration as selected by the computer program. The output also identifies at each state (or iteration) the component selected for redundancy. Included in the output data are:

a. The state (or iteration)

b. The identification of the redundant component by number (The components are numbered consecutively.)

c. The total number of redundancies of that particular component

d. The total weight of the satellite system

e. The time t used to calculate MMD (which is a function of the expendables aboard)

f. The MMD of the total system

g. The reliability of the total system at the time t

4. PAYLOAD STATE CONCEPT AND AVAILABILITY

A payload is in an "up" state if all its modules, including its housekeeping modules and experiment modules, are operational or in non-failed states. A payload is in a "down" state if any of its modules is in a failed state. In the SAMS computer program, the state change time points of each payload are stored for each Monte Carlo trial. A Monte Carlo trial is a typical mission simulation starting at time zero and ending at some prescribed time. For each Monte Carlo trial, SAMS keeps track of the state change time points and sums the individual up times from a prescribed "availability start time" (TA) to the simulation end time (TM). For each Monte Carlo trial we have P/L I:

$$\bar{U}_i = \text{mean up time in trial i for P/L I} \tag{1}$$

$$= \frac{I}{NU_i(I)} \sum_{j=1}^{NU_i(I)} U_{ij}(I)$$

where

$U_{ij}(I)$ = j^{th} up time in trial i for P/L I

$NU_i(I)$ = number of up times in trial i for P/L I

$A_i(I)$ = availability of P/L I in trial i

$$= \frac{1}{(TM-TA)} \sum_{j=1}^{NU_i(I)} U_{ij}(I)$$

It should be understood that the up time or availability of P/L I is the up time or availability of the function associated with P/L I. A deployment and retrieval of P/L I is assumed to take place in the same orbital position.

The above statistics apply to a single Monte Carlo trial. In order to obtain statistical confidence, a number of trials must be made to obtain the ensemble outputs.

Comparable measures can also be defined for total system availability where the total number of payloads is considered, and the availability can be generally defined in terms of a subset of payloads. For the total system, the quantities analogous to equations (1) and (2) can be an output.

5. LAUNCH AND MODULE USAGE RATES

The previous subsection was concerned with the availability performance parameters A (availability) and U (mean up time). Equally or more important are the cost impact parameters of launch and module usage rates. For each Monte Carlo trial i, SAMS accumulates the number of

failures over the simulation interval TM resulting from anomalous events in module J of P/L I. The ensemble statistics for the Monte Carlo trials are:

$$NF(I,J) = \text{Number of failures resulting from anomalous events in module J of P/L I} \tag{3}$$

$$= \frac{1}{MC} \sum_{i=1}^{MC} NF_i(I,J)$$

$$NF_i(I,J) = \text{Number of failures resulting from anomalous events in module J of P/L I for } i^{th} \text{ trial.}$$

$$SF(I,J) = \text{Standard deviation of } NF_i(I,J) \tag{4}$$

$$= \frac{1}{MC}\left[\sum_{i=1}^{MC} NF_i^2(I,J) - NF^2(I,J)\right]^{1/2}$$

A typical Shuttle launch can involve the replacement and retrieval of an entire payload and also the servicing of multiple modules within a payload or payloads. For each Monte Carlo trial i, SAMS accumulates the number of launches resulting from anomalous events of any module. The total number of launches resulting from anomalous events is summarized by the ensemble statistics.

6. PROGRAM INPUTS AND LIFE SIMULATION METHODS

These paragraphs describe the reliability inputs to the simulation and the methods of simulating module life. Also described are the following general inputs to the simulation:

NS = Number of payloads to be established and maintained on-orbit

TM = Simulation time of mission interval

NF = Number of failures required for a Shuttle launch

MC = Monte Carlo sample size

RSTRT = Random number start value

t_{L1} = Launch reaction time for on-orbit repair launch

t_{L2} = Launch reaction time for ground-refurbishment launch

t_m = Repair time on-orbit for each payload

R_L = Launch reliability (Failure means a second attempt is made

R_D = Docking reliability per payload (Docking failure aborts the repair mission for subsequent payloads but does not require an immediate new launch if there are no unrepaired failures.)

R_r = Repair reliability on-orbit per payload (Unsuccessful repair requires an immediate new launch.)

The payload module reliabilities are the primary input to the simulation because they determine the launch rate of the Shuttle. The module reliabilities are modeled by Weibull functions, because the Weibull is the simplest generalization of the exponential and has the flexibility of considering decreasing as well as increasing hazards. The warning functions are also modeled as Weibull. In the simulation, any failure triggers a launch, and it is assumed that warnings are not involved in the launch decision; however, warnings which have occurred at the launch decision point have priority in determining the orbit service load. The order of module selection for service of occurred warnings is ordinarily based on a first event - first served basis, but it is possible to modify this order by some weighting scheme. A weighting scheme is the reciprocal mean life of the module where a module with a large mean life can be pushed back in the priority order. The result is that a module pushed back in the order may not be loaded until the next Shuttle launch. The simulation currently assumes that the first payload visited is the failed payload, and subsequent payload service ordering is based on the order prescribed by the separation angle; i.e., the exact warning ordering or predictive ordering scheme used in determining the Shuttle/Tug load is generally overridden by orbit geometry consideration in order

for the flight to carry as much load as possible. Accumulated failures, however, are serviced first.* Also, all service called for by the priority ordered loading for a particular payload (failed and warned modules) are serviced in one visit.

7. MODULE LOADING (SERVICE) ORDER

The previous subsection described the methods of simulating or generating warning and failure times for a selected module. This subsection describes the methods of selecting the order of modules to be loaded on the Shuttle/Tug. As mentioned previously, a launch is based on a failure and this failure (first failure first if there is an accumulation) is to be serviced first before other modules selected for servicing. Assuming additional modules can be loaded, servicing of occurred warnings is usually the next consideration on a first event - first served basis. After loading the Shuttle with these modules, assuming that additional modules can be loaded, the selection is next based on a predictive mode.

The priority order may be summarized as follows:

a. Load failed module replacement first. If there are accumulated failures, load the modules on a first failed - first served basis.

b. After completing the loading of failed module replacements, select further modules from occurred warnings on a first event - first served basis or on some modified ordering of occurred warnings.

c. If still further loading is possible, select service modules based on a predictive mode.

The logic implemented to select the module loading order for predicted failure replacements is that certain modules can be tagged to be selected first (in order of tagging) in the predictive mode. Other modules can be untagged and never selected in the predictive mode. After selecting

*The assumption of servicing failures first may be changed at a later date. This assumption is made in view of the fact that outage occurs only for failures and not for warning and also because the relaxation of this assumption opens the door to a tremendous range of ordering methods.

the tagged modules and assuming that still further modules can be selected for service, SAMS selects the modules according to the following options:

a. Random or no failure information mode. Order unwarned and unfailed modules on the basis of a random ordering where all possible orderings are equally likely.

b. Perfect failure prediction or PFP mode. Other modules in increasing order of actual generated failure times. The failure time of each module is accurately predicted.

c. Conditional life mode. Order modules in increasing order according to their 50 percentile conditional life value In the computation, the operational time of the module taken into account.

The PFP mode considers the accurately generated failure times whereas the conditional life mode considers the failure distribution function. None of the three predictive modes considered is concerned with warning; thus, warnings only enter into the simulation as events which have occurred prior to a launch-triggering failure. This is appropriate, because prediction of a warning is of less concern than the honoring of a warning that has occurred and thus, when a prediction is necessary, the prediction should be for the failure event rather than for the warning event.

D. LAUNCH AND PAYLOAD SERVICE RATES

The expected number of launches that were computed in the previous subsection is used to determine the Shuttle launch and payload service rates. This data is required to evaluate the trip charges and the ground-refurbishment and on-orbit maintenance charges.

The Shuttle launch cost can be shared with other missions on the flight since the Orbiter and Tug can perform other operations such as deployment, servicing, and retrieval and return of payloads. The amount of sharing will depend on the performance capability. In the case of the Shuttle/Tug, the performance is limited. Performance was not considered in the case of

the Shuttle only, because there was adequate deployment and retrieval performance capability for the payloads considered in the study.

For the Shuttle/Tug, the performance capability was based on the values provided in Reference 4 which are listed as follows for payload placement into synchronous orbit:

Round Trip	= 1360 kg (3000 lb)
Retrieve and Return	= 1910 kg (4210 lb)
Deployment	= 3620 kg (7980 lb)

These performance limits were used to determine the trip sharing based on payload weights. For the payload weight range of $W \leq 1360$ kg (3000 lb), the expendable payloads were assumed to share the trip with another payload, but a ground-refurbishable payload was not shared because the Tug is performing a round-trip operation by deploying a payload and retrieving and returning another payload. For payloads in the weight range of 1360 kg (3000 lb) $< W \leq 3620$ kg (7980 lb), the expendable payloads were assumed to be a dedicated flight. For ground-refurbishable payloads, the payload weight range was limited to 1360 kg (3000 lb) $< W \leq 1910$ kg (4210 lb). Two dedicated trips are involved, because one trip is required for deployment and a second trip is required to retrieve the ground-refurbishable payload. The ground-refurbishable payload weight cannot exceed this limit in order for the payload to be returned for refurbishment.

For on-orbit maintenance payloads, the Shuttle/Tug must be capable of servicing non-space-replaceable components and on-orbit replaceable modules. The repair of non-space-replaceable components must be accomplished by returning the payload to ground. Thus, the payload weights for one trip charge must be limited to $W \leq 1360$ kg (3000 lb). The repair of on-orbit replaceable modules was assumed to be accomplished by a servicing unit adapted to the Tug to house the modules and to remotely changeout modules. This servicing unit ard maneuvering impulse to service the second payload were assumed to weigh 454 kg (1000 lb) which leaves 908 kg (2000 lb)

for modules. The module loading capability, if two payloads are serviced, is $W_m \leq 454$ kg (1000 lb). If the module weight exceeds this limit, the maintenance mission would service only one payload and the added maneuvering impulse would not be required. The module loading capability for this case is then 454 kg (1000 lb) $< W_m \leq 1130$ kg (2500 lb). If the module weight exceeds this limit, the maintenance mission will require two trip changes.

With these performance constraints, the equations to determine the launch rates for the Shuttle/Tug and payload including the servicing were then formulated. These launch rates and payload service rate equations are summarized in Table 2-2 for three payload types. For all of these cases, it was assumed that the last payload in the replacement/service series remains on-orbit after the mission has been completed and that the last ground-refurbishment payload or modules are returned to avoid debris in space but are not refurbished on the ground. Also, one spare payload was included for the ground-refurbishable payload, one set of replacement modules was included for an on-orbit maintenance payload, and no spare hardware was included for expendable payloads. These equations are limited to a one-payload system; i.e., they do not apply to a multiple-payload system on-orbit. The equations were programmed as a subroutine in the payload program cost model to provide equivalent launch rates for trip charges and payload service rates for ground-refurbishment and on-orbit maintenance charges.

E. PAYLOAD PROGRAM COST ESTIMATING

1. INTRODUCTION

From past payload studies, a comprehensive payload program cost model has evolved for analyzing a total space program composed of numerous individual payload (satellite) programs. This basic cost model has been simplified and reprogrammed for a remote terminal computer system that allows individual payload programs to be modified and analyzed quickly and

Table 2-2. Shuttle/Tug Launch and Payload Rate Equations

	Shuttle/Tug		Payload		
	Total Launch	Equivalent Launch	New	Refurbish.	Maintenance
Expendable					
W≤1360 kg (3000 lb)	NE	1/2NE	NE	0	0
1360 kg (3000 lb)< W≤3620 kg (7980 lb)	NE	NE	NE	0	0
Ground Refurbishable					
W≤1360 kg (3000 lb)	NE_R	NE_R-1/2	2	NE_R-2	0
1360 kg (3000 lb)< W≤1910 kg (4210 lb)	$2NE_R$-1	$2NE_R$-1	2	NE_R-2	0
On-Orbit Maintenance					
W≤1360 kg (3000 lb)					
eW_m≤454 kg (1000 lb)	$N(E_m+E_R)$	$N(1/2E_m+E_R)$	2	NE_R	$(NE_m-2)e$
eW_m>454 kg (1000 lb)	$N(E_m+E_R)$	$N(E_m+E_R)$-1/2	2	NE_R	$(NE_m-2)e$
1360 kg (3000 lb)< W≤1910 kg (4210 lb)					
eW_m≤1130 kg (2500 lb)	NE_m+2NE_R	NE_m+2NE_R	2	NE_R	$(NE_m-2)e$
eW_m>1130 kg (2500 lb)	$2NE_m+2NE_R$	$2NE_m+2NE_R$-1	2	NE_R	$(NE_m-2)e$

Notes:
- E = Total expected number of launches
- E_R = Expected number of launches for ground refurbishment
- E_m = Expected number of launches for on-orbit maintenance
- e = Ratio of number of modules replaced to total number of modules in payload
- N = Program duration

efficiently. Subroutines have been added to the program so that the lowest total program cost can be automatically identified from among a large number of alternative design and program strategies. The purpose of this subsection is to:

a. Provide a description of the basic cost model
b. Define the inputs it requires
c. Discuss the output of the model

2. PAYLOAD PROGRAM COST MODEL

The model is composed of two major sections; one computes satellite costs and the other computes launch vehicle cost. For the Shuttle/Tug, launch vehicle cost is defined as direct cost per flight and includes such items as expendable drop tanks, pro-rata solid motor hardware, propellants (solid and liquid), recovery, refurbishment, spares, and launch site vehicle maintenance and flight support.

Payload cost is defined as all effort required to design, develop, and manufacture satellites and support them during launch and orbital operation. Typically, a satellite program is divided into RDT&E (nonrecurring), investment (recurring), and operations (recurring) cost categories, and the model treats each cost estimate accordingly. RDT&E covers design and development, investment includes procurement of satellite hardware, and operations covers support during and after launch. In cases where reuse through ground refurbishment or on-orbit servicing is considered, the operations category also includes repair and refurbishment.

The basic payload model calculated RDT&E and unit cost from payload data that are automatically applied to cost-estimating relationships (CERs) stored in the program. Launch vehicle cost information is a direct input. Based on payload and launch vehicle quantities, satellite unit and program total costs are calculated and printed for each alternative.

The payload computer model has been modified to aid in selecting lowest-cost alternatives from a wide range of performance and program

variables. The categories of variables that affect cost are type of satellite (expendable, ground refurbishable, on-orbit maintainable, or Sortie), number of replaceable modules per payload, MMD (which is reflected in weight differences from the baseline payload), number of modules replaced per visit, numerous cost sensitivity candidates (such as launch costs, ground refurbishment rates, or on-orbit servicing rates), and program lifetimes.

Inputs are obtained from the Spacecraft Synthesis and the SAMS programs, which provide payload and schedule (quantity) data. These data were manually inputted to the payload model; however, the model is organized so that all estimates covering all of the variable mentioned above are calculated, and the lowest-cost alternative is identified for each category.

3. COST MODEL INPUTS AND ASSUMPTIONS

The performance data and the payload descriptive and schedule (quantity) information required for operating the cost model are set forth in Table 2-3. Descriptions of all inputs and the necessary assumptions that relate to their use are presented in this section. In addition, the following basic assumptions apply to the cost estimates generated for this study:

a. Constant 1973 dollars used

b. No low-cost designs considered; i.e., those based on relaxed weight and volume constraints

c. Standardized modules not considered; therefore, no savings accrue from shared developments or quantity production

d. Complete RDT&E costs included for each satellite alternative, except that, for reusable satellites, one equivalent set of payload parts and 75 percent of mission equipment RDT&E are used to estimate effects of shortened R&D time and cost when the Shuttle system is used

e. Sortie mode user charge is $0.7 million per flight based on MSFC data; no Sortie RDT&E is charged. Only one set of mission equipment is used

f. No payload infant mortality is included; however, all other reliability effects are included in yearly expected flights

Table 2-3. Satellite Cost Estimate - Basic Input Information

Input Item Description	Examples (or Remarks)
Title	(Six-digit number for identification)
Satellite type	Expendable, on-orbit maintainable, ground reusable, and Sortie
Structure weight Electrical power weight Communications weight Stability and control weight, dry Stability and control weight, consum. Propulsion weight, inerts Propulsion weight, propellants Mission equipment weight	(Baseline expendable weights by subsystem--when satellite is different from baseline, then non-baseline weights must also be an input.)
Mission equipment type	Communication, earth resources, etc.
Initial electrical power	Watts
Propulsion, type	Solid or liquid, if subsystem needed
Propulsion, total impulse, lb/sec	If subsystem needed
Orbit altitude	Low, sync., planetary
Number of satellites in system	(Required in orbit for system operation)
Constant year dollars	1973
Launch vehicle type	Shuttle or Shuttle/Tug
Schedules (quantity)	(Quantity based on yearly expected flight requirements
Structure type	Exostructure or endostructure
Stability type	3-axis or spin
Refurbish rate (for ground refurbishment)	39 percent
Repair rate	25 percent for one equivalent satellite's worth of modules
No. of replaceable modules/ payload	2 to 19 in a satellite
No. of modules replaced/visit	One to all per flight
Launch vehicle cost ($ million)	Shuttle is 10.3; with Tug, 11.2

g. Ground refurbishment based on replacing non-repairables, e.g., batteries and propellants; repair of failed items; and complete production acceptance test of all replaced parts and modules

h. One RDT&E for program duration

i. Total program cost based on slice of continuing program, i.e., residual value is not included

a. Title and Satellite Type

For identification purposes, a six-digit number is used as a title for each satellite program alternative. Each digit represents a different category of alternatives:

1. The first digit represents the type of satellite; i.e., expendable, ground refurbishable, on-orbit maintainable, or Sortie. Expendable refers to typical current designs that assume expendable satellites. Ground refurbishable means that current technology and design procedures are used but that they are modified to allow for reuse. On-orbit maintainable payloads have space-replaceable modules and non-space-replaceable components. Sortie payloads assume that mission equipment is flown in a Sortie Lab at low orbits for short periods of time and then returned to earth. The digit will be 1, 2, 3, or 4 depending on type. (Similarly, the digits below will vary from 1 to 9 depending on the number of alternatives in each category.)

2. The second digit defines the number of replaceable modules in a particular reusable satellite design. The number of modules will vary from 2 to more than 20.

3. The third digit defines the MMD, which varies from 1 to 4.5 years for the cases considered in this study.

4. The fourth digit defines the number of modules replaced per visit, depending on the number of modules in a particular design. The modules replaced will vary from one to all.

5. The fifth digit defines repair rates for ground or on-orbit serviced alternatives.

6. The sixth digit defines launch vehicle cost (trip charges).

Program lifetimes are varied for all of the above alternatives but are not identified by a separate digit.

b. Subsystem Weights

Expendable weights are an input, because the cost-estimating relationships (CERs) are based on typical expendable satellites. Factors are applied to the baseline cost estimates to provide for ground-refurbishable or on-orbit maintainable designs. The factors are based on differences in weight from baseline subsystems and thus require additional weight input data.

c. Mission Equipment Type

Ten types of mission equipment are identified in the cost model:

1. Communications
2. Navigation
3. Earth Resources
4. Meteorology
5. Surveillance
6. Technology Applications
7. Astronomy
8. Plasma Physics
9. Solar Observation
10. Planetary

For a particular estimate, the most appropriate category must be selected from the list.

d. Initial Electrical Power

Input requires initial output of the electrical subsystem to be given in watts.

e. Propulsion Type and Total Impulse

A propulsion system is not normally required by most satellites; however, such a requirement should not be confused with the attitude control propulsion, which is part of stability and control. The Type refers to the

propellant used, either solid or liquid. Total Impulse in kg/sec (lb/sec) is also a required input when a propulsion subsystem is needed.

f. Orbit Altitude

The orbital altitude at which the satellite operates is a required input. Three categories are considered: low, synchronous, and escape (planetary).

g. Number of Satellites in System

Many programs require more than one satellite to be in orbit during operations; quantity required is a needed input.

h. Constant Year Dollars

Cost estimates reflect constant dollars, as desired by a particular analysis, and provision is made for such an input.

i. Launch Vehicle Type

The cost of launch vehicles is set in the program; however, the Shuttle or the Shuttle/Tug must be identified by an input.

j. Schedules (Quantity)

The SAMS program generates yearly expected launches for each type of satellite and for each strategy. These yearly expected values, together with the program lifetimes which are used in the computer to calculate quantities of payloads launched and number of launch vehicle flights required, were described previously.

k. Structure Type

Type of structure refers either to endo structure (associated with compact, spin-stabilized satellites) or exostructure (associated with less compact, three-axis-stabilized satellites).

l. Stability Type

The stability systems may be either spin or three-axis controlled.

m. Refurbishment Rate

The refurbishment rate does not apply unless the satellite is ground refurbishable. The rate, when applied to the average unit cost, gives a cost per flight of repairing and refurbishing a satellite that has been returned from orbit. Nominal rate is 39 percent; for sensitivity analyses, 50 percent and 30 percent are used. Sortie is three percent per flight applied to mission equipment only.

n. Repair Rate

Repair rate applies only when a satellite is designed for on-orbit maintenance. The computer program calculates servicing cost based on the number of replaceable modules carried to orbit, the number of modules that make up the satellite, and the average repair rate. The nominal rate is 25 percent; for sensitivity calculations, 30 percent and 20 percent are used.

o. Number of Replaceable Modules Per Satellite (and Per Visit)

Depending on the servicing strategy, one or more modules will be carried to an orbiting satellite when a failure occurs or a warning system demands a repair flight.

p. Launch Vehicle Cost

Any launch vehicle can be considered; however, this study uses the Shuttle, Shuttle/Tug, or Shuttle/Sortie Lab; costs are $10.3 million based on NASA/MSFC inputs, $11.2 million, or $11.0 million, respectively. The Tug rate is $0.9 million, and the Sortie Lab rate is $0.7 million.

4. REDUNDANCY AND MODULARITY

Previous satellite studies on extended sign life (Air Force Programs 191 and 777) have examined the effect on subsystem weight and cost

of extending satellite lifetimes through redundancy. Factors were developed that equated subsystem weight increases with cost increases for redundant equipment. As part of the current study, additional work has been done on the DSP satellite, and the results have been integrated with the previous studies. As a consequence, adjustments were made to the communications and mission equipment subsystems; however, no change to electrical or stability and control subsystems occurred.

The actual calculation is based on comparing a particular design with the baseline, establishing a weight change factor, determining the cost change factor from the weight factor, and applying the cost factor to the baseline cost estimate. The Payload Program Model automatically computes the required factors and applies them to the RDT&E and unit cost estimates.

Similar methods are used by the model to give effect to modularity for on-orbit serviced designs; however, structure cost is based solely on cost estimating relationships (CERs). In addition, the number of modules in a particular design influences the cost of stability and control. Again, the computer model makes all of the required calculations based on the inputs discussed earlier.

5. CER ANALYSIS

Over the past several years, satellite cost data have been collected in an effort to verify the accuracy of cost-estimating relationships used in various payload-related studies. These data were assembled and compared with CER estimates; the conclusion was reached that the CERs gave reasonably acceptable estimates. For this study, a more detailed analysis was conducted to gauge the effect on cost estimates of possible changes in the CERs. The method used for this analysis was to measure the approximate change in estimates of cost for a typical satellite. An average from the sample data for each subsystem was used as proxy for weight and performance inputs. Costs were then calculated from the original CERs and from

possible revised CERs,* and a comparison of the results was made as shown below

Subsystem	Weight	Total Cost in $ Millions	
		Original CERs	Revised* CERs
Structure	102 kg (225 lb)	10.2	9.7
Electrical	300 (watt)**	11.2	16.3
TT&C	57 kg (125 lb)	26.5	21.2
Stab. and Control	68 kg (150 lb)	27.1	25.7
TOTAL		75.0	72.9

From the above set of figures, it can be seen that the difference between the previous CERs and a possible revision would result in a change of less than three percent. Based on this analysis, it appears that a major revision in payload CERs (including the substantial amount of effort such an undertaking would require) is not critical for this study.

*The CERs have not actually been revised; an approximation was used based on a best-fit representation of the actual subsystem cost data collected for all satellites in the sample.

**The electrical CER is based on power rather than weight.

3. DESIGN ANALYSIS

A. INTRODUCTION

Design analysis was performed on the Large Space Telescope (LST) and the Shuttle-Launched Defense Support Program (SLDSP) to develop quantitative cost data. These costs were developed to reflect the effects of reliability, repair strategy, and payload type. This type of payload data required the parts counts of each subsystem, the overall design concept, and weight estimates.

For both payload programs studied, the reliability analysis and conceptual designs were performed simultaneously because of the study schedule. This approach was satisfactory since the study is to provide parametric cost-trend data and not a point-design analysis. The reliability analyses examined payload configurations ranging from single string to maximum MMD. The conceptual design task investigated varying numbers of modules in the payloads and different payload types. The design effort emphasized on-orbit maintenance concepts for both low-altitude and high-altitude operations. The low- and high-altitude servicing is performed by the Orbiter and Tug, respectively, where each has different servicing approaches. From these independent efforts, the design factors related to reliability, modularity, and payload type were developed as input to the mathematical model.

The costing effort in the mathematical model computed systematically the total program cost for a range of payload data and expected launch rates that were provided from the design task and reliability analysis. Cost data dependent on the various parameters were then varied determined as a function of variations in program duration, repair cost, and trip charges to evaluate their sensitivity to the various payload parameters. As indicated earlier, the effort was not aimed at a point design but rather to find payload cost trends.

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B. LARGE SPACE TELESCOPE DESIGN ANALYSIS

The Large Space Telescope (LST) was selected as the low-altitude payload for the design analysis. The baseline version of the LST spacecraft is illustrated in Figure 3-1 and described in Reference 5. The baseline LST is man-tended and non-modularized. It is composed of three functional elements entitled optical telescope assembly (OTA), scientific instrument package (SIP), and support systems module (SSM). The OTA includes the optics, forward structure, sunshade, baffles, and systems for telescope thermal control alignment, and focus. The SIP consists of the scientific instruments, supporting optics, sensors, electronics, positioning mechanisms, and the structure located at the telescope focal region. Typical instruments are field cameras, UV and IR spectrographs, and polarimeters. The SSM includes the aft structure and the subsystems required to support the OTA/SIP. The SSM also provides a docking structure for on-orbit servicing or retrieval by the Space Shuttle. The SSM is designed to be pressurizable to provide a shirt-sleeve environment for an astronaut during equipment servicing.

1. RELIABILITY MODEL

The reliability block diagrams were constructed for the scientific instrument package, optical telescope assembly, and support systems modules from the data supplied in Reference 5. Failure rates, duty cycles, and weights of individual components were determined from the LST preliminary and Phase A technical reports. In some instances where failure rate information was not available but a reliability number was shown for some time t, equivalent failure rates were determined for the equipment using the exponential assumption. Equipment considered to be unessential to successful operations were deleted in the optimization study. Figures 3-2 through 3-6 shows the reliability block diagram of the LST. The following assumptions were made:

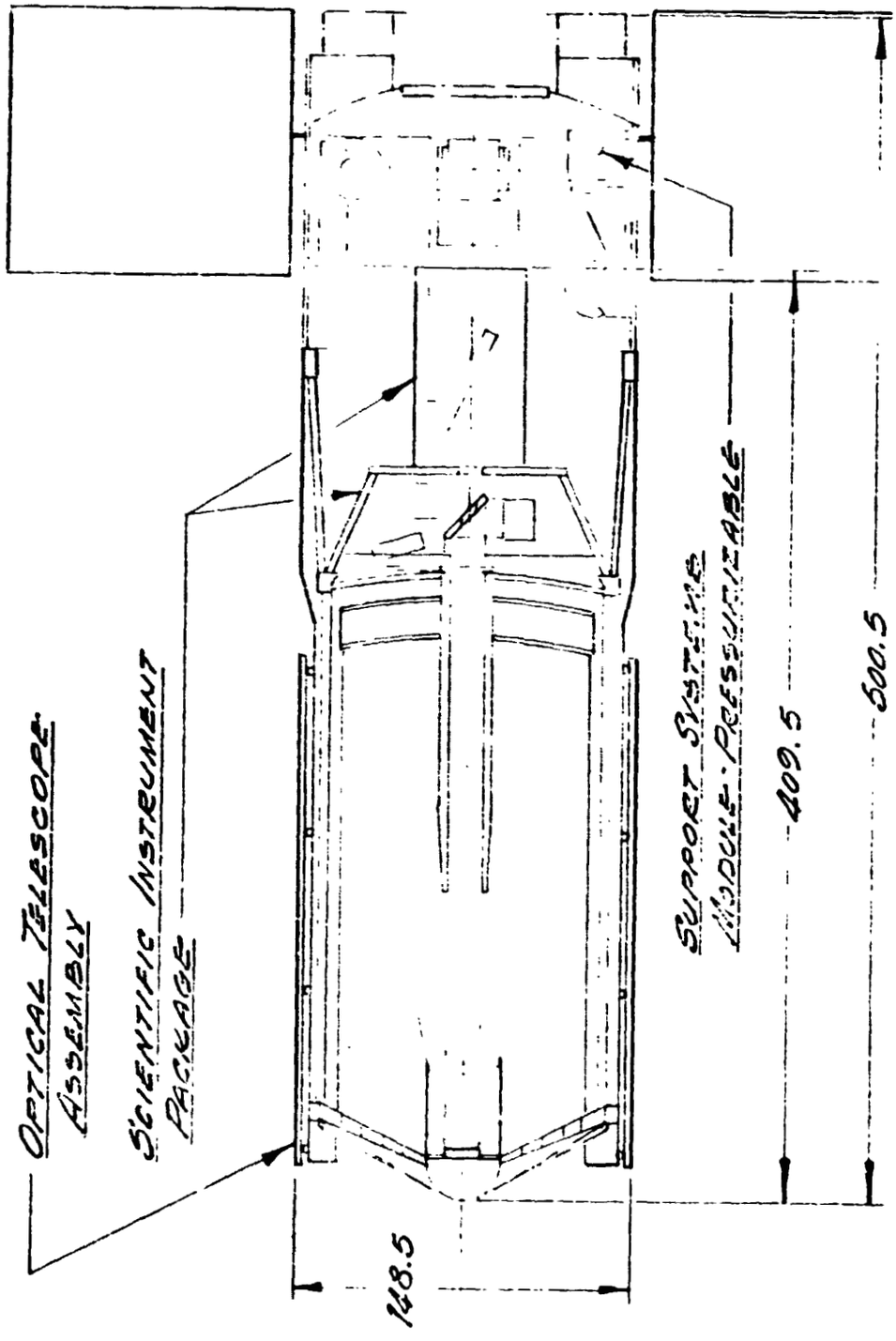


Figure 3-1. MSFC Large Space Telescope (LST)

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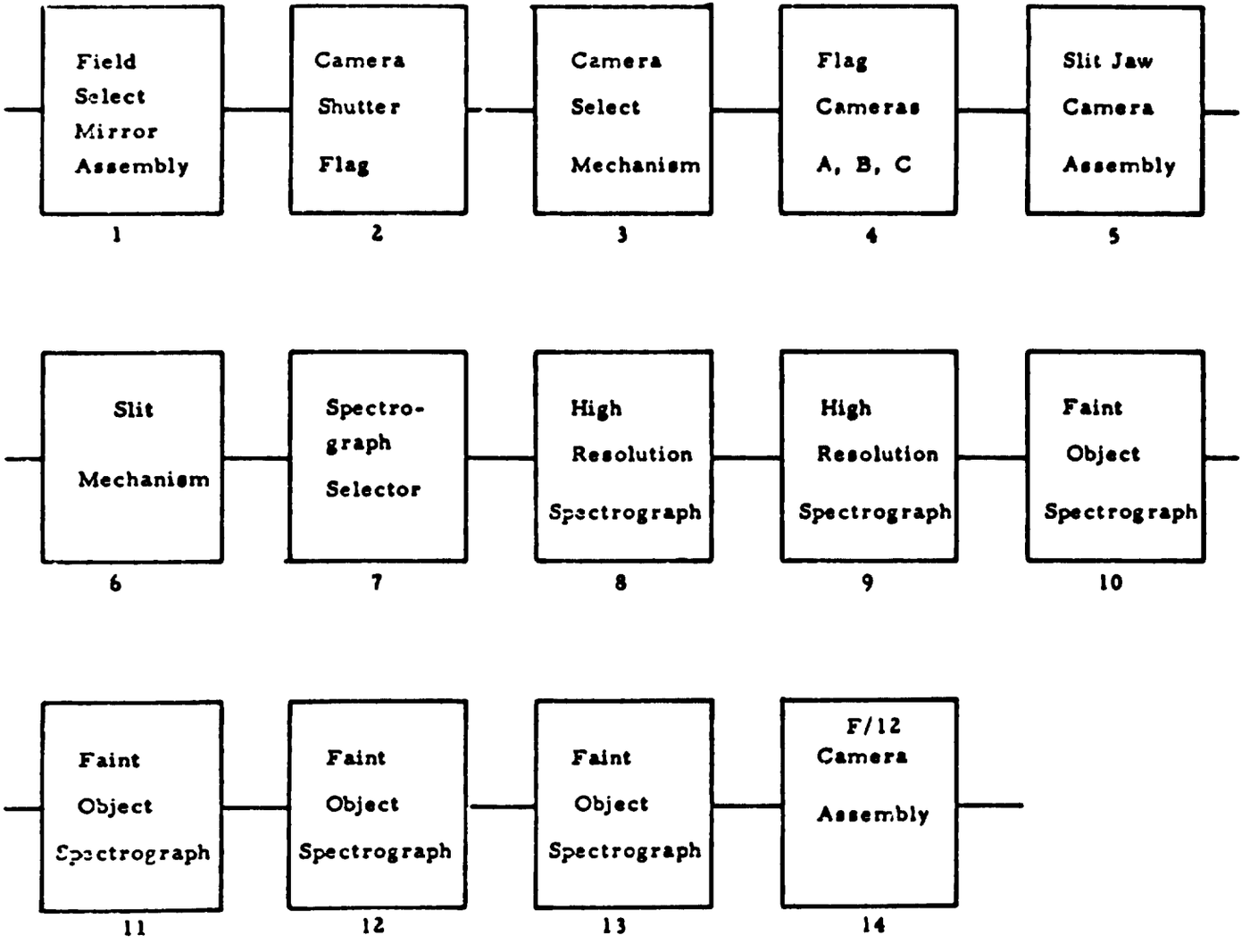


Figure 3-2. LST Scientific Instrument Package (SIP) Reliability Model (No Redundancy)

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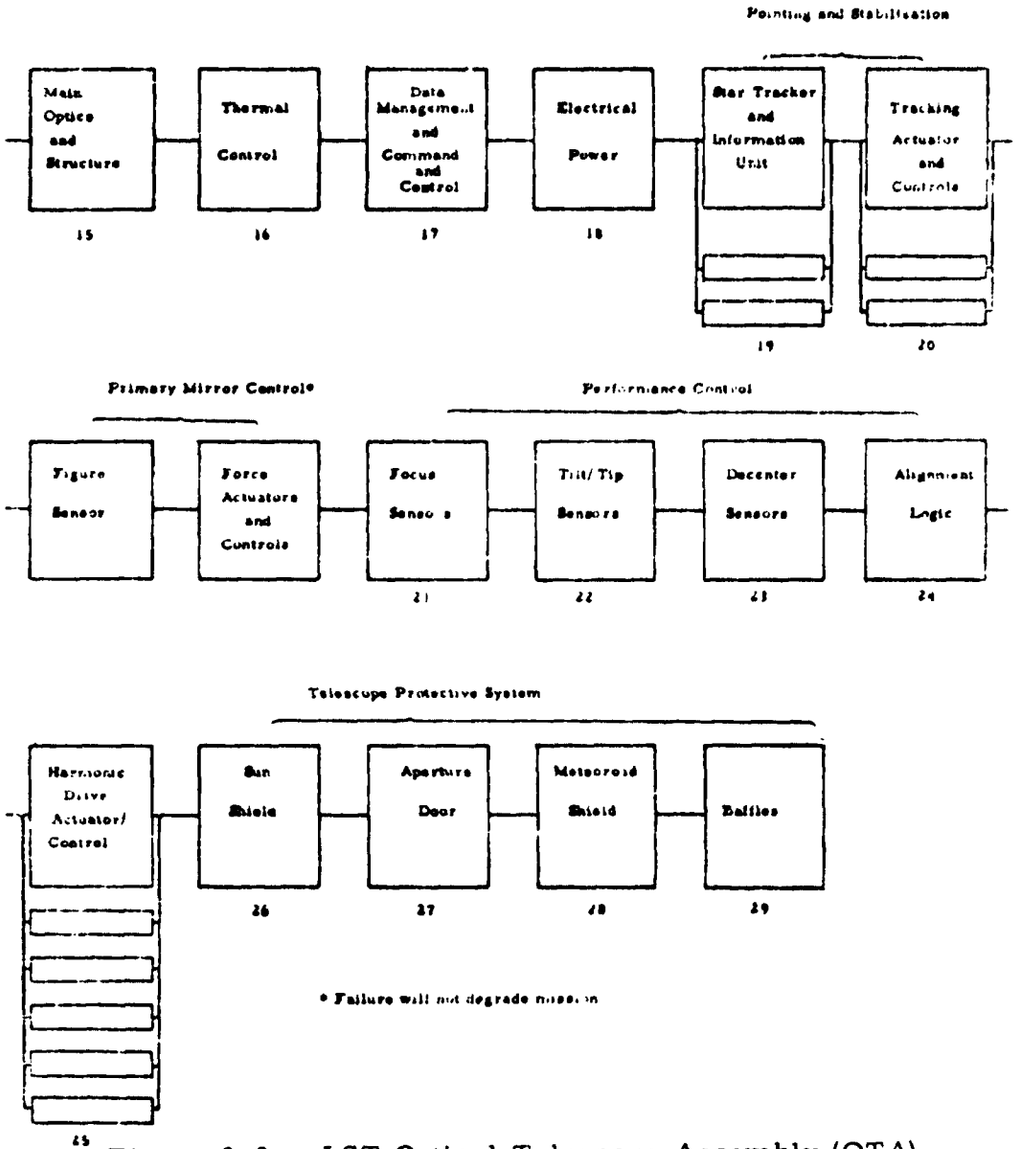


Figure 3-3. LST Optical Telescope Assembly (OTA) Reliability Model (No Redundancy)

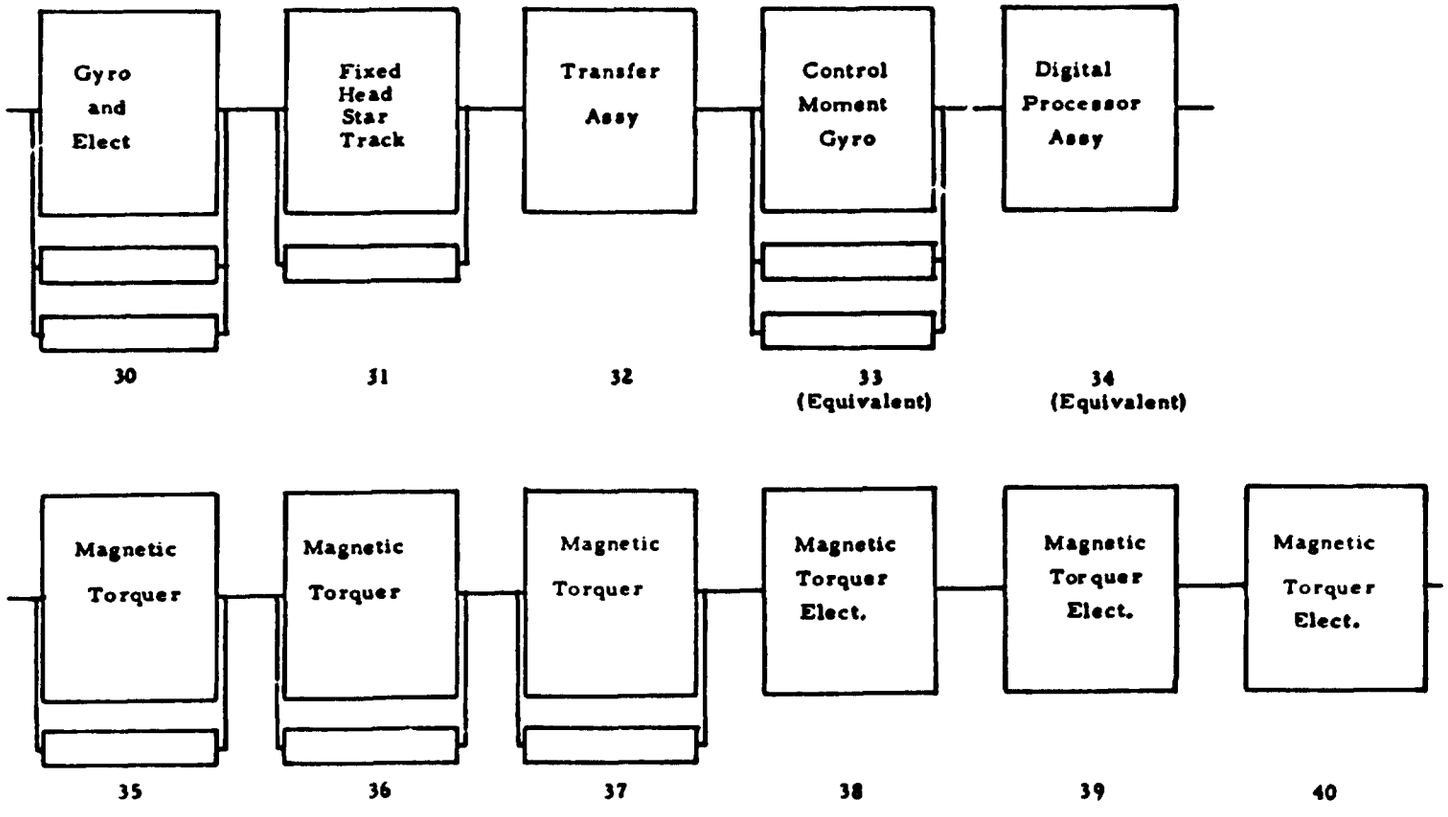


Figure 3-4. LST Attitude Control Subsystem Reliability Model

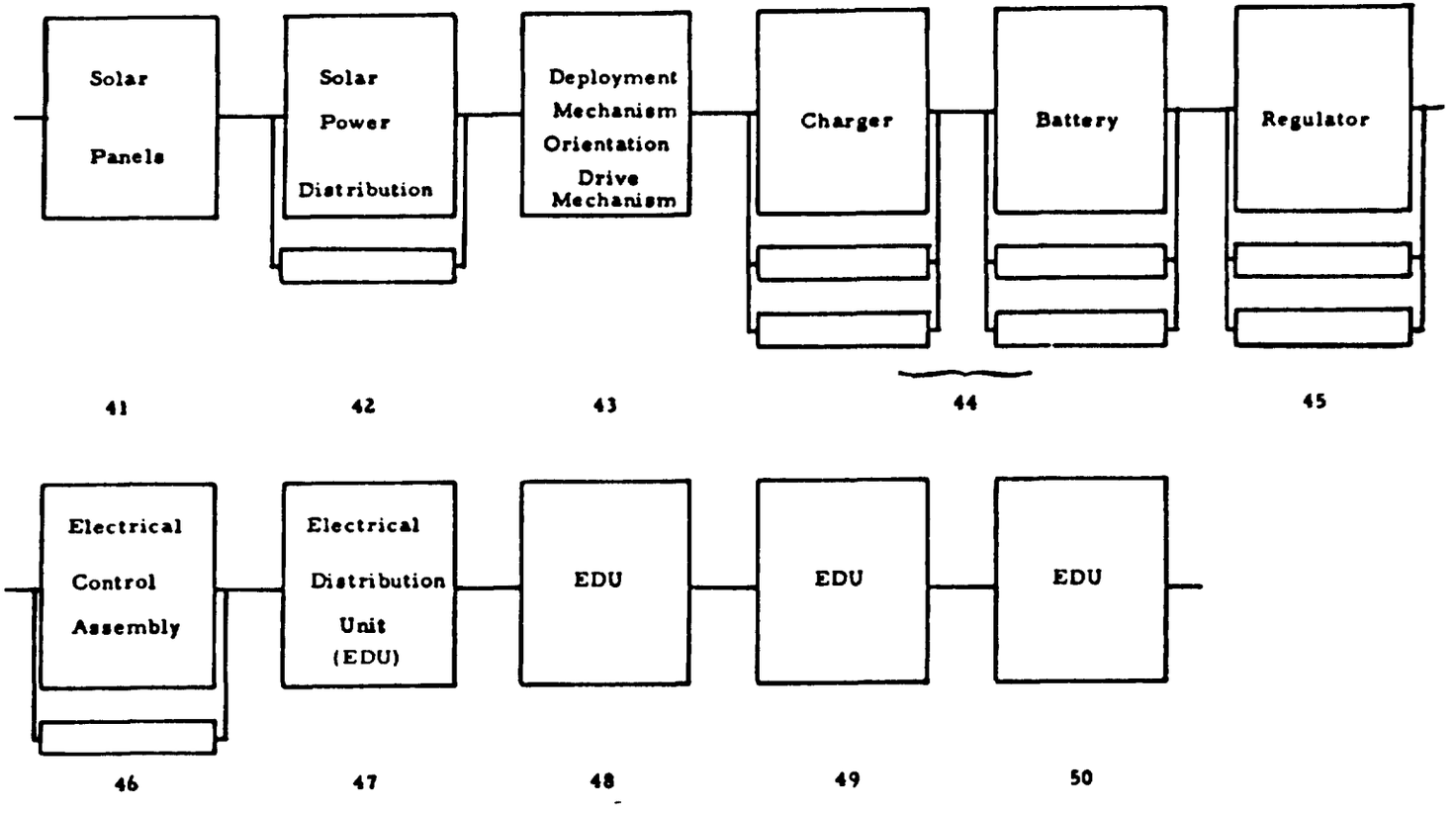


Figure 3-5. LST Electrical Power Reliability Model

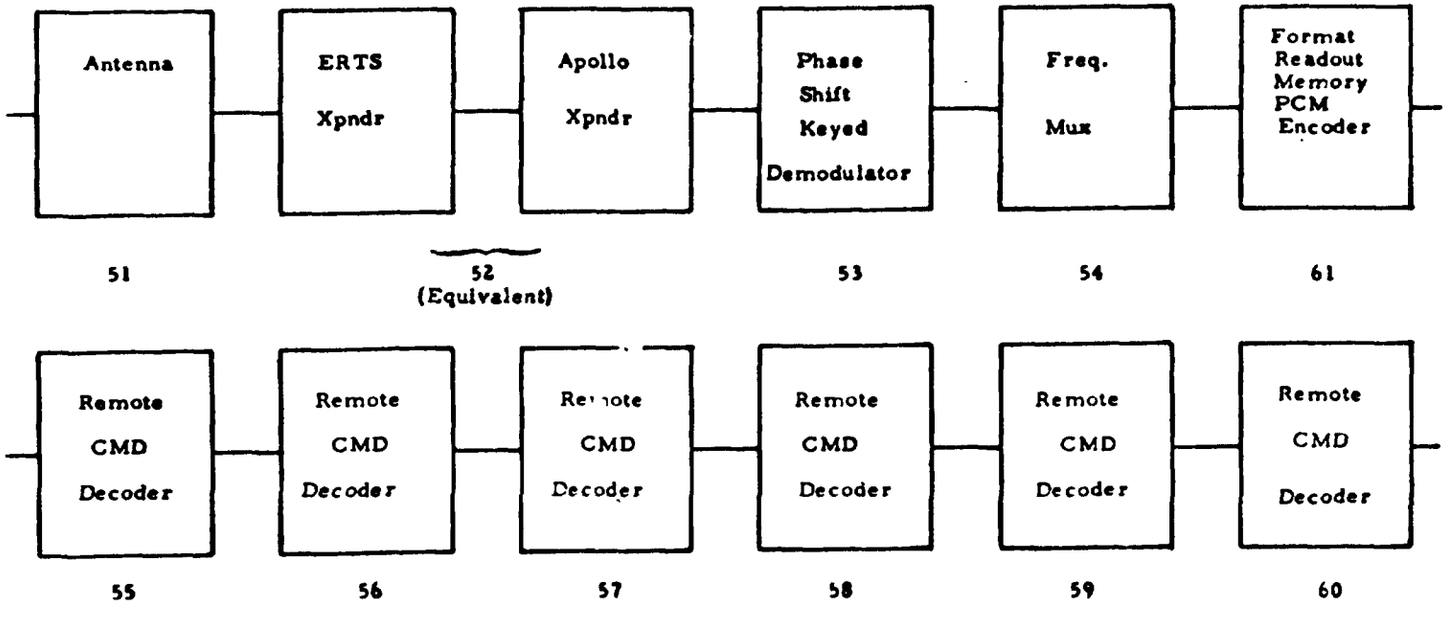


Figure 3-6. LST Communication and Data Handling Reliability Model

a. All components were assumed not to reach their wearout points. Failure rates for standby units were assumed to be equal to one tenth the active failure rates. The switching mechanisms are accounted for in the standby redundancies.

b. The control moment gyros (CMG) in the attitude control subsystem of the SSM are binomial redundant requiring a minimum of three. Each gyro is composed of several components in active and standby redundancies. A reliability curve was generated for a single CMG and was approximated by an exponential curve with an equivalent failure rate. This was required due to the limitations of SYSOPT.

c. The digital processor assembly in the attitude control subsystem of the SSM contains the power supply, the control processor unit, and the memory modules in standby redundancies. Due to lack of detailed weight data, the digital processor assembly was treated as a single component with internal redundancies. A reliability curve was generated for the component, and a normal curve (μ = 61,320 hr, σ = 24,528 hr) was used to approximate the reliability. Redundancy of a component described by the normal failure distribution implies a standby failure rate equal to zero.

d. The ERTS and Apollo transponders on the communications and data handling subsystem of the SSM have a backup mode of operation for the transmitters. Using a 30-percent duty cycle for the transmitters, a reliability curve was generated for the two transponders combined. An exponentail reliability curve was fitted to the original curve and the equivalent failure rate used for the failure rate of the combined transponders.

e. For the optimization, no redundancies were added to the SIP and the OTA.

The block diagrams with expendables for a year's mission time and stripped of unnecessary redundancies were used as the starting system configuration. Expendables were added in amounts equal to three-month intervals with the weight of 10.4 kg (22.93 lb) being divided equally between the propellant and the accompanying hardware.

The SYSOPT program that was described previously was used to determine the parts count as the MMD was extended. The system weight dependency on the MMD as the MMD was varied from the single string case to a maximum of 1.7 years is shown in Figure 3-7. From this data, 0.5, 1.0, and 1.5 year MMDs were selected for detailed tradeoff analysis. At

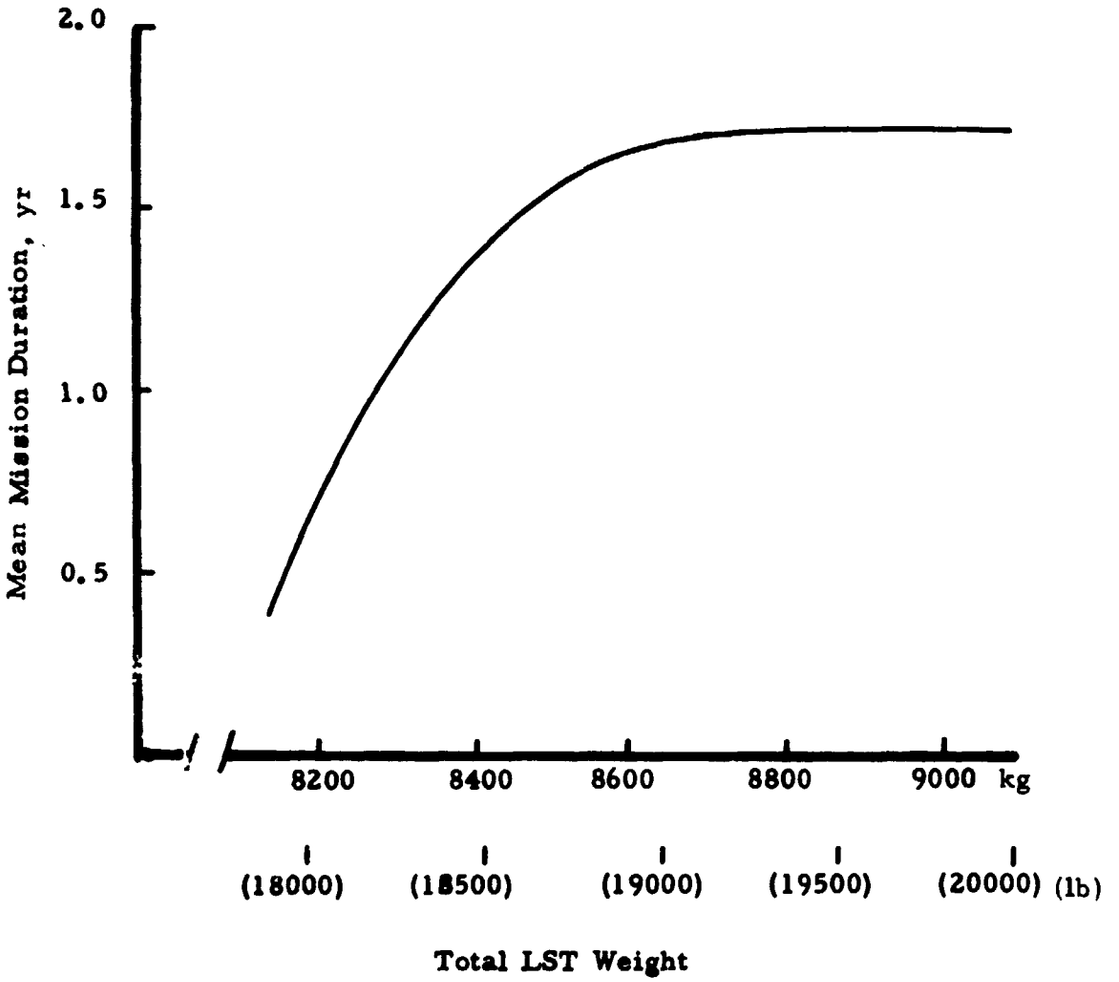


Figure 3-7. LST Weight Growth With MMD

these MMDs, the parts count and weight are shown in Tables 3-1, 3-2, and 3-3 for the housekeeping subsystems. The component failure rates and the MSFC baseline configuration parts count are also included. The MSFC baseline MMD was estimated to be 1.25 years. The parts list for the mission equipment is shown in Tables 3-4 and 3-5; this equipment was not made redundant in this study.

2. CONCEPTUAL MODULARIZED DESIGN

Conceptual design was performed to identify and evaluate feasible modularized configurations of a Shuttle-launched Large Space Telescope (LST). The study provided conceptual configurations of the LST that were suitable for module replacement by means of the Orbiter-based remote manipulator and/or a hard-docked servicing module. The principal design ground rules were as follows:

a. The modular LST design should be amenable to equipment replacement by means of the Orbiter operator-controlled remote manipulator (Ref. 6).

b. In addition to the spacecraft subsystems complement, the LST instrumentation (Scientific Instrument Package) is to be replaceable in orbit.

c. All modular LST configurations are to be unmanned and unpressurized.

d. Telescope energy may be time-shared by the scientific instruments.

e. Subsystems and/or components of long-life expectancy; e.g. spacecraft structure and solar arrays should not require replacement.

A number of modularized LST concepts were identified. These are illustrated in Figures 3-8 through 3-12. Several of the concepts are suitable for use with a DSP-type servicing module, and the remainder are specifically configured for equipment servicing by the Orbiter remote manipulator. Examination of the master equipment list (Tables 3-4 and 3-5) indicates that the LST subsystems and instrumentation can be contained in the volume provided by a vehicle 4.42 m (14.5 ft) in diameter and approximately 2.44 m

Table 3-1. LST Stabilization and Control Subsystem Failure Rate, Parts List, and Weight

Component	Failure Rate	Unit Weight		Parts Count				Weight							
				B/L	0.5	1.0	1.5	B/L		0.5 yr		1.0 yr		1.5 yr	
	(10^{-9})	kg	lb					kg	lb	kg	lb	kg	lb	kg	lb
Reference Gyro Assy	15,000	1.8	4	6	4	6	8	10.5	23	7.3	16	10.7	24	14.5	32
Fixed Head Star Tracker (and Shades)	4,411	7.0	15.5	3	2	3	4	21.1	47	14.1	31	21.3	47	28.2	62
Transfer Assembly	5,957	4.7	10.3	2	1	2	2	9.4	21	4.7	10	9.2	21	9.2	21
Control Moment Gyros and Digital Electronics Assembly	869	80.8	178	4	3	3	3	324.0	712	243.0	534	243.0	534	243.0	534
Digital Processor Assy	μ=61320 hours σ=24528 hours	6.2	13.7	1	1	1	2	6.2	14	6.2	14	6.2	14	12.5	27
Magnetic Torquers	40	23.6	52	6	6	6	6	141.0	310	141.0	310	141.0	310	141.0	310
Magnetic Torquer Electronics	702	0.4	0.85	6	3	6	6	2.3	5	1.1	3	2.3	5	2.3	5
3-Axis Magnetometer		3.0	6.5	2	2	2	2	5.9	13	5.9	13	5.9	13	5.9	13
Coarse Sun Sensors		0.1	0.2	5	5	5	5	0.4	1	0.4	1	0.4	1	0.4	1
Lines, Valves and Regulator Set		43.9	96.7					43.9	97	43.9	97	43.9	97	43.9	97
RCS Electronics		7.3	16	2	2	2	2	14.5	32	14.5	32	14.5	32	14.5	32
RCS Thruster Module		2.0	4.5	4	4	4	4	8.2	18	8.2	18	8.2	18	8.2	18
RCS Tank		32.4	71.3	1	1	1	1	32.4	71	33.5	74	54.5	120	91.0	200
Expendables (GN_2)								19.5	43	20.6	46	41.7	92	78.2	172
TOTAL								639	1405	544	1198	603	1326	693	1524

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Table 3-2. LST Electrical Power Subsystem Failure Rate, Parts List, and Weight

Component	Failure Rate	Unit Weight		Parts Count				Weight							
				B/L	0.5	1.0	1.5	B/L		0.5 yr		1.0 yr		1.5 yr	
	(10^{-9})	kg	lb					kg	lb	kg	lb	kg	lb	kg	lb
Solar Panels		93.2	205	2	2	2	2	186.2	410	186.2	410	186.2	410	186.2	410
Solar Power Distribution		5.5	12	2	2	2	2	10.9	24	10.9	24	10.9	24	10.9	24
Solar Panel Mechanisms		58.2	128	2	2	2	2	116.2	256	116.2	256	116.2	256	116.2	256
Charge/Battery	37,870	29.2	64.3	6	4	5	8	175.2	386	116.8	257	146.0	322	233.3	514
Regulator	11,100	3.5	7.8	6	4	5	7	21.4	47	14.2	31	17.7	39	24.8	55
Electrical Control Assembly	3,643	7.3	16	2	2	2	3	14.5	32	14.5	32	14.5	32	21.8	48
Electrical Distribution Units	536	1.1	2.5	8	4	4	8	9.1	20	4.5	10	4.5	10	9.1	20
Cabling and Connect								113.5	250	113.5	250	113.5	250	113.5	250
Manned Support and Maintenance Equipment								38.6	85	-	-	-	-	-	-
TOTAL								686	1510	577	1270	610	1343	716	1577

Table 3-3. LST Communication and Data Management Subsystem Failure Rate, Parts List, and Weight

Component	Failure Rate	Unit Weight		Parts Count				Weight							
				B/L	0.5	1.0	1.5	B/L		0.5 yr		1.0 yr		1.5 yr	
	(10^{-9})	kg	lb					kg	lb	kg	lb	kg	lb	kg	lb
Antenna	130	0.9	2	2	2	2	2	1.8	4	1.8	4	1.8	4	1.8	4
Apollo Transponder	2,500	14.5	32	1	1	1	2	14.5	32	14.5	32	14.5	32	29.1	64
ERTS Transponder		10.9	24	1	1	1	2	10.9	24	10.9	24	10.9	24	21.8	48
Phase Shift Keyed Demondulator	1,788	0.5	1	2	2	2	3	0.9	2	0.9	2	0.9	2	1.4	3
Data Control Unit	1,147 DC = .3	1.8	4	2	1	1	2	3.6	8	1.8	4	1.8	4	3.6	8
Remote Command Decoders	1,800	0.4	0.8	16	12	12	12	5.8	12.8	4.4	9.6	4.4	9.6	4.4	9.6
Format Generators, Command Processor and Memory	14,280 DC = .3	5.0	10.9	2	1	2	3	9.9	21.7	5.0	10.9	9.9	21.7	14.8	32.6
Communication and Data Handling Switch		0.5	1	2				0.9	2	0.9	2	0.9	2	0.9	2
Digital Acquisition Unit		0.1	0.25	28				3.2	7	3.2	7	3.2	7	3.2	7
Tape Control		1.3	3	2				2.7	6	2.7	6	2.7	6	2.7	6
Tape Recorders		5.5	12	3				16.4	36	16.4	36	16.4	36	16.4	36
Clock		0.5	1	1				0.5	1	0.5	1	0.5	1	0.5	1
TOTAL								71	157	63	139	68	149	101	221

Table 3-4. LST Scientific Instrument Package Subsystem Failure Rate, Parts List, and Weight

Component	Failure Rate	Unit Weight		Parts Count
	(10^{-9})	kg	lb	B/L
Field Select Mirror Assembly	0.114	0.50	1.0	1
Camera Shutter Flag	60.0	0.5	1.0	1
Camera Select Mechanism	8.0	22	47.9	1
F/96 Cameras, A, B, C	11700	217	478	1
Slit Jaw Camera Assembly	3331	48	105.2	1
Slit Mechanism	48.0	5	10.5	1
Spectrograph Selector	9.4	8	17.4	1
High Resolution Spectrograph 204-333 km (110-180 nmi)	1100	63	137.6	1
High Resolution Spectrograph 333-648 km (180-350 nmi)	1100	63	137.6	1
Faint Object Spectrograph 204-407 km (110-220 nmi)	4900	64	141.5	1
Faint Object Spectrograph 407-1222 km (220-660 nmi)	3125	49	108.3	1
Faint Object Spectrograph 1222-1852 km (660-1000 nmi)	1321	48	104.8	1
Faint Object Spectrograph 1852-9260 km (1000-5000 nmi)	183	50	110.5	1
F/12 Camera Assy	1450	71	157	1
Miscellaneous (Including Structure)		296	650.3	
TOTAL		1005	2[illegible].3	

Table 3-5. LST Optical Telescope Assembly Subsystem Failure Rate, Parts List, and Weight

Component	Failure Rate	Unit Weight		Parts Count
	(10^{-9})	kg	lb	B/L
Structure		788	1734	1
Optics	22.8	2083	4589	1
Thermal Control	102.8	103	226	1
Data Management	137.1	3	7	1
Electrical Power Distribution	194.3	34	75	1
Fine Guidance Assy	9,060 DC= .5	117	257	3
Secondary Mirror Tracking Actuators	5,850 DC= .5	30	65	3
Focus Sensors	3,102 DC=.02	13	28	1
Angle Sensor	3,108 DC=.02	9	20	2
Decenter Sensor	3,102 DC=.02	16	36	2
Diagnostics	500 DC=.02	24	52	1
Secondary Mirror Alignment Actuators	10,500 DC=.02	66	145	6
Figure Sensor	3,505 DC=.02	17	38	1
Primary Mirror Force Transducers	29,000 DC=.02	17	38	24
Light Shield	480.5	3440	758	1
Aperture Door Assembly	744.4	99	217	1
Meteoroid Shield	2.0	532	1171	1
Baffles	6.0	292	644	1
TOTAL		4587	10098	

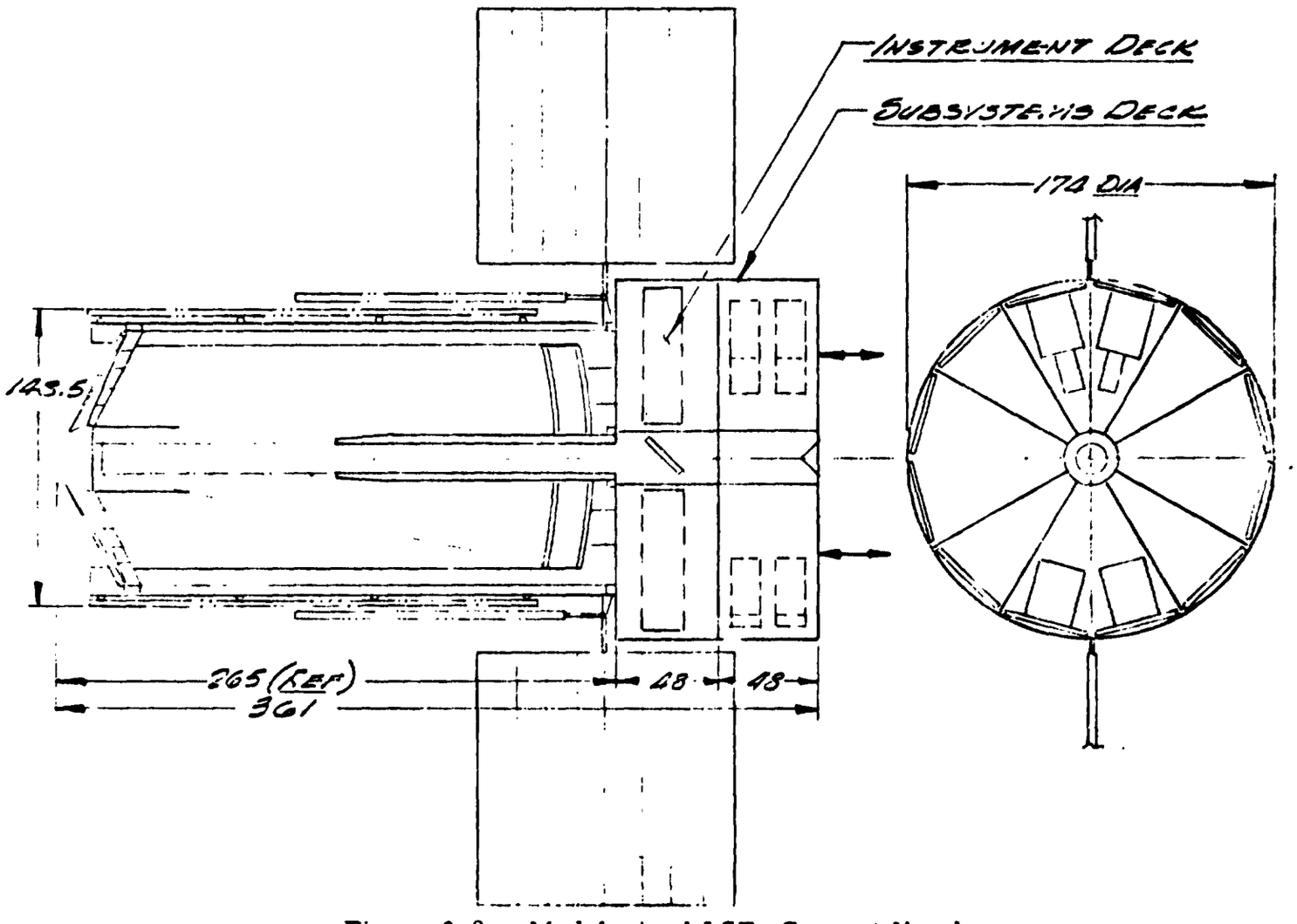


Figure 3-8. Modularized LST, Concept No. 1

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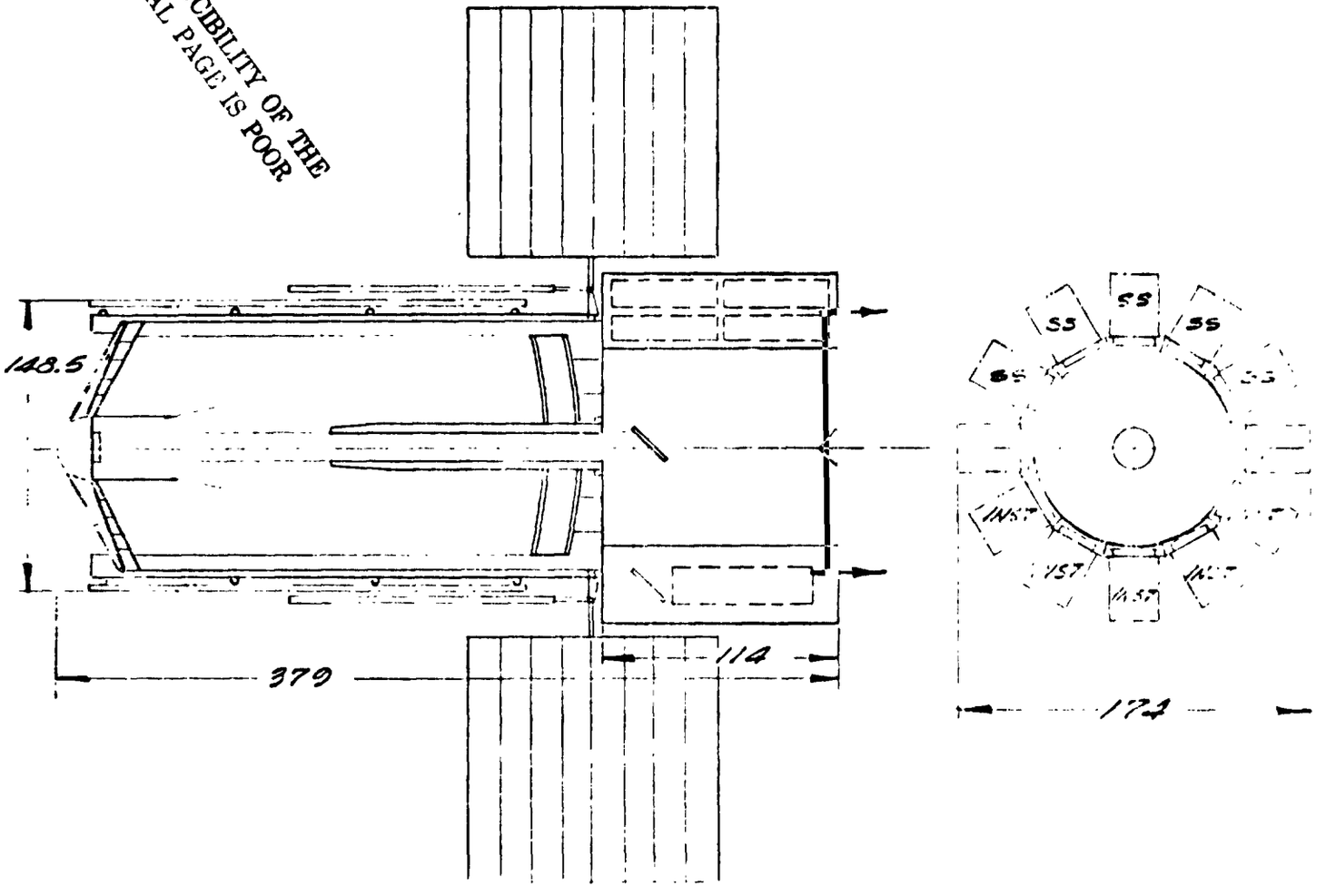


Figure 3-9. Modularized LST, Concept No. 2

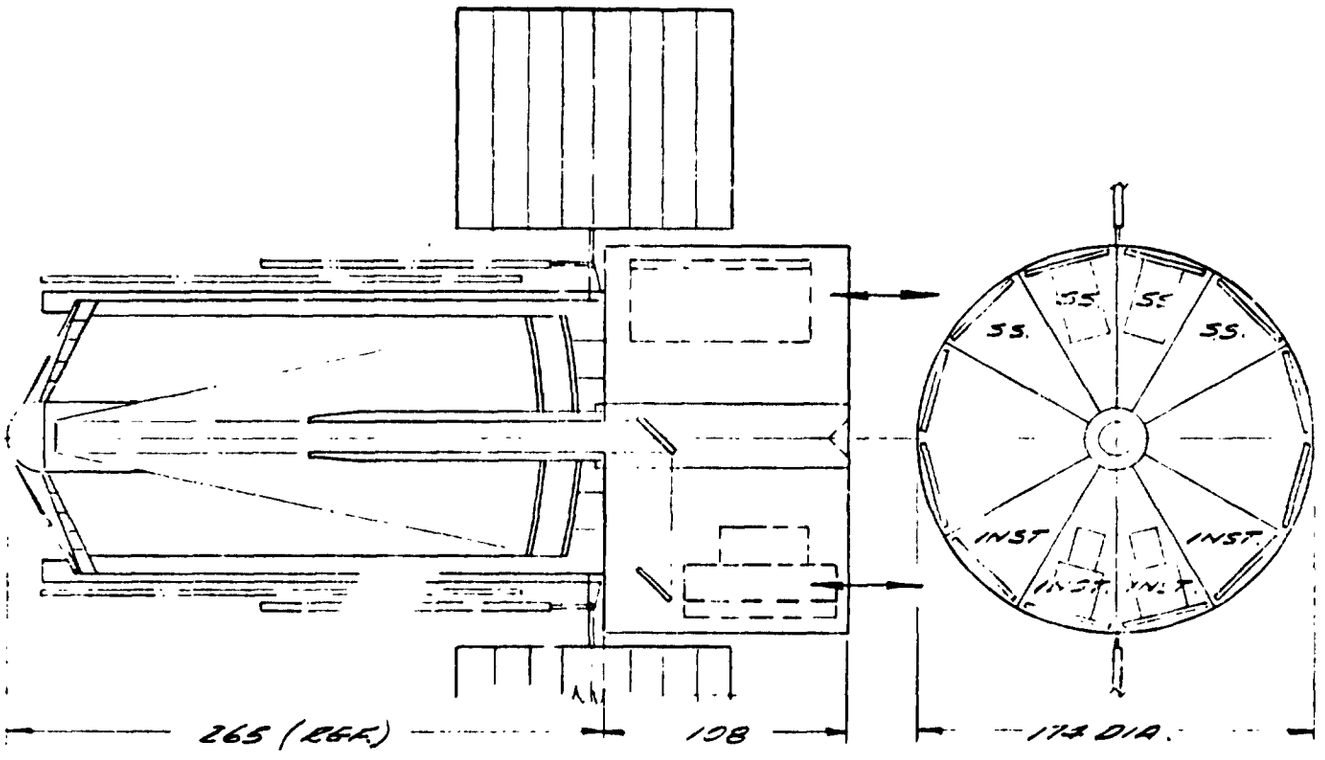


Figure 3-10. Modularized LST, Concept No. 3

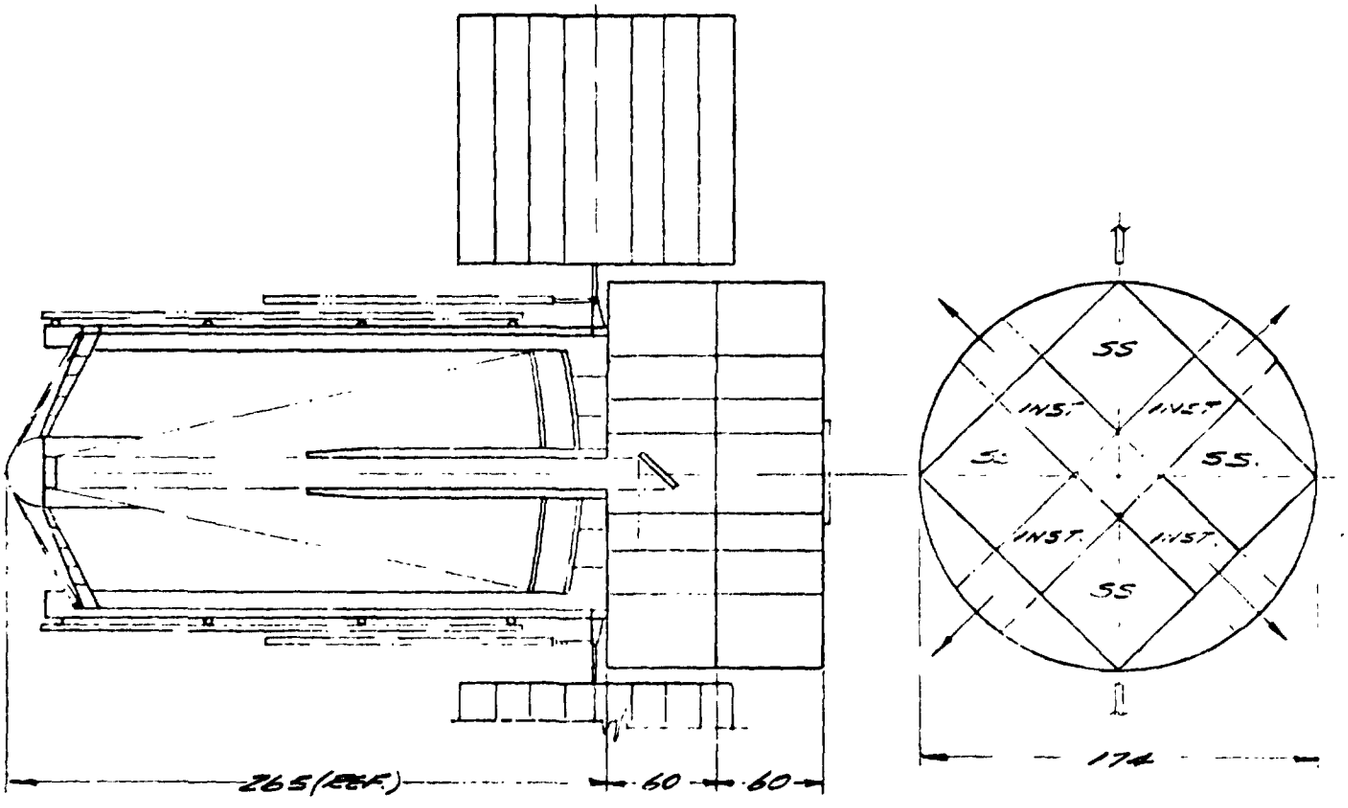


Figure 3-11. Modularized LST, Concept No. 4

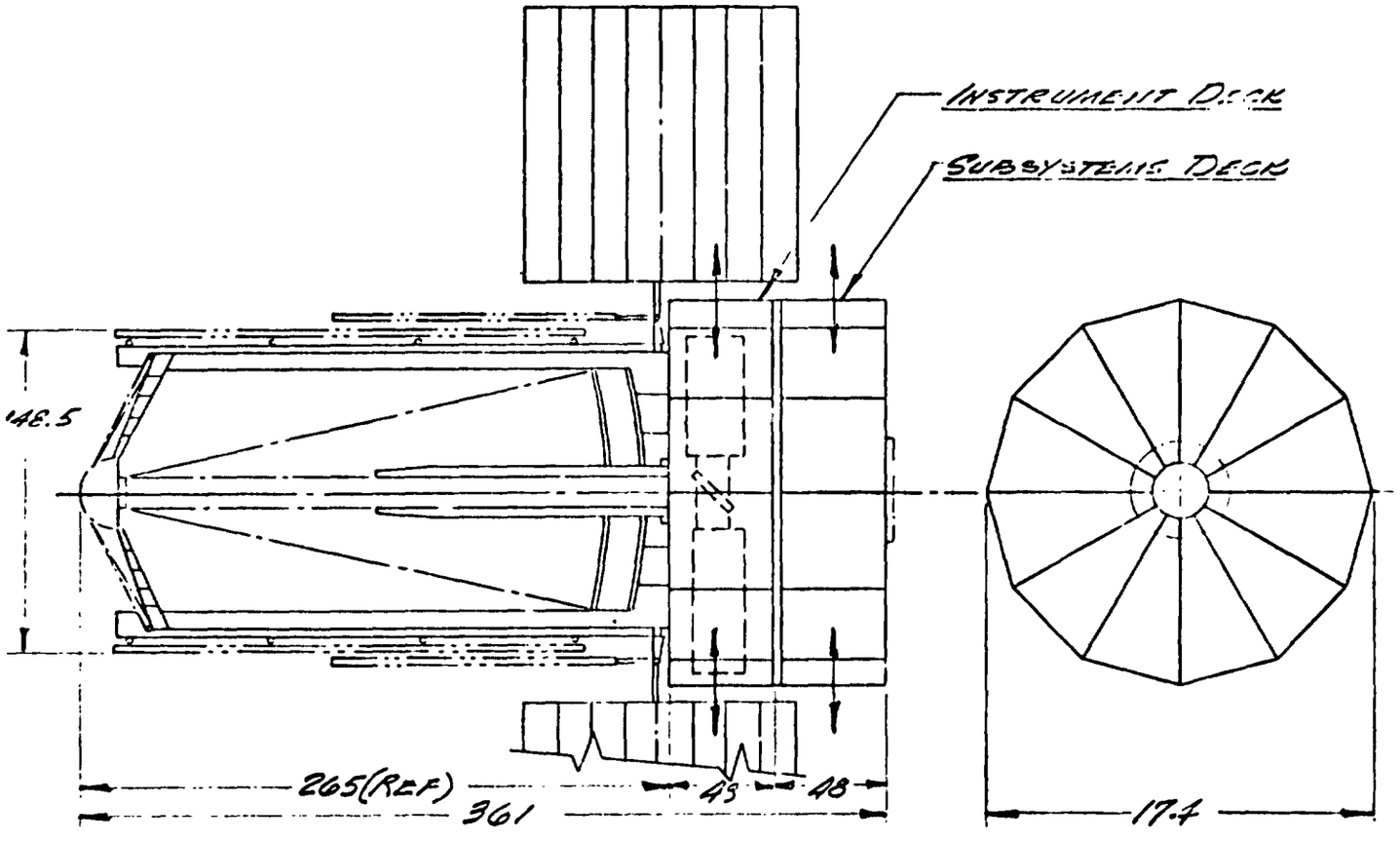


Figure 3-12. Modularized LST, Concept No. 5

(8 ft) in length. Figure 3-8 illustrates a concept wherein the LST instrumentation and subsystems are contained in two separate decks 1.22 m (4 ft) in length. This concept could be utilized with either a SLDSP-type servicing vehicle or the Orbiter manipulator. In this concept, the equipment in the aftmost deck would have to be removed first in order to gain access to the instruments in the forward deck. This procedure would require the servicing vehicle to be designed with a two-position translation probe in order to change-out the spacecraft equipment and would also necessitate the use of two rotating elements or decks on the servicing vehicle in order to accommodate equipment and service the spacecraft. For these reasons, the concept of Figure 3-8 was rejected.

The concept of Figure 3-9 utilizes one long deck and installs the equipment in the axial (parallel to the vehicle longitudinal axis) direction. In this design, the equipment modules are located tangentially around the periphery of the spacecraft. The design of Figure 3-9 is directly applicable to refurbishment by a hard-docked service vehicle. This concept could be used with the remote manipulator.

Figure 3-10 illustrates another concept that does not require a two-position translation probe if used with a hard-docked servicing vehicle and which is adaptable to use with the manipulator. It was not considered to be a strong candidate concept due to its apparent weight disadvantage compared with the concept of Figure 3-11 and lack of noticeable design improvement.

Figure 3-11 depicts a concept that could be used with the manipulator but which is incompatible with the equipment replacement mode of the hard-docked service vehicle. The basic (square) shape exhibited by this concept results in some wasted cargo bay volume. This is particularly objectionable in the case of a spacecraft such as the LST wherein some of the instrumentation is on the order of 1.83 m (6 ft) in length. For these reasons, the concept of Figure 3-11 was not considered further.

Figure 3-12 illustrates a concept that is well adapted for use with the manipulator as shown in Figure 3-13. A detailed discussion of this configuration and a comparison of this concept and the design of Figure 3-9 are presented in the following sections.

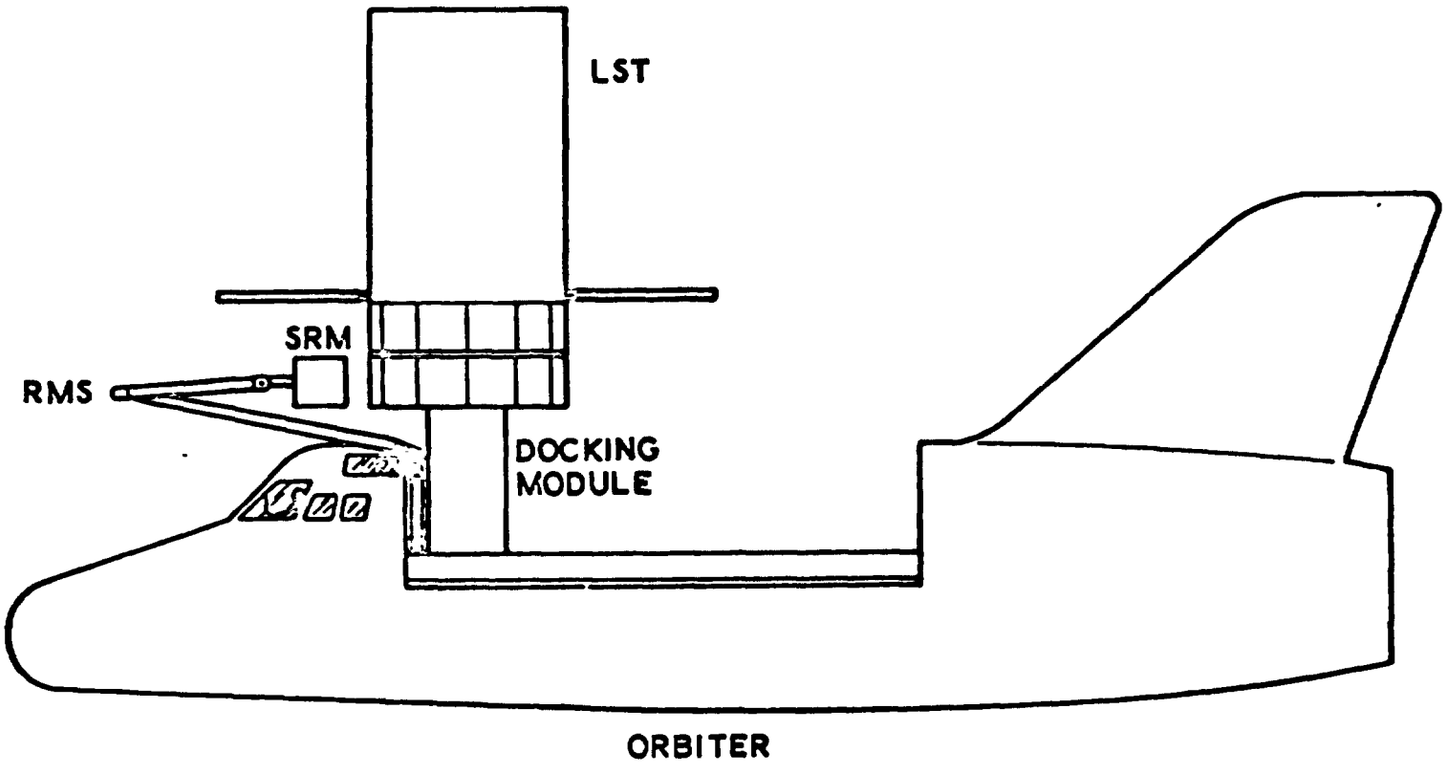


Figure 3-13. Swing Translation to Replace Module for On-Orbit Maintenance Using Orbiter RMS

a. Comparison of Modular Concepts

The two most promising modular LST payload concepts identified above are shown in Figures 3-9 and 3-12. The concept of Figure 3-9 illustrates a configuration that is amenable to equipment replacement by means of a hard-docked service module. In this mode, the equipment modules would be removed and replaced in a (longitudinal) direction parallel to the spacecraft axis. The service module or compartment (containing the replacement modules) could be carried in the aft end of the Orbiter cargo bay and elevated to a suitable position for docking with the LST after opening of the cargo bay doors.

Although the Orbiter-borne manipulator has the kinematic capability (travel and degrees of freedom) to replace equipment modules by movement in the payload axial direction, the length of a module that can be replaced in the axial direction by means of the manipulator is at present severely limited by the proximity of a payload to the Orbiter fuselage when the payload is docked to the cargo bay extensible docking port [0.762-m(30-in.) clearance] (Ref. 7). If it is assumed that this limitation may not ultimately exist (due either to future revision of the extensible docking port/Orbiter geometry or to incorporation of a more versatile serviceable-payload docking arrangement), the configuration of Figure 3-9 can be considered to be a candidate concept suitable for servicing by either a hard-docked service module or by the manipulator.

The concept of Figure 3-9 stores the scientific instruments longitudinally around the periphery of the payload structure. The modules would be at least 1.83 m (6 ft) in length to accommodate the instrumentation. In this case, the module support structure would basically be in the form of a ring or shell approximately 2.44 or 2.74 m (8 or 9 ft) in diameter with the subsystems and instrumentation mounted external to the shell. The modules are designed to be removed in the axial (longitudinal) direction.

The concept of Figure 3-12 is designed to permit radially outboard removal of the modules by means of the Orbiter manipulator. The instrumentation length is accommodated in the radial direction. In this concept,

the module support structure would occupy the full 4.42 m (14.5 ft) diameter of the allowable payload envelope.

The design of Figure 3-9 probably reflects a payload weight that is lower than that of the "radial" concept (Figure 3-12) due to the smaller diameter of the payload support structure. This weight advantage may not, however, represent a critical selection criterion for a low-earth orbit payload due to the large low-altitude payload launch weight capability of the Orbiter. In this design, the instrumentation receives its input energy from the LST telescope via auxiliary folding mirrors which, together with the instrument complement, are external to the spacecraft structure. This design therefore exhibits some potential light-sealing and contamination problems that are not present with the concept of Figure 3-12 wherein the instrumentation and subsystems are totally contained within the external structure. A second set of fold mirrors is required for the design of Figure 3-9 in order to reflect the (already reflected) light energy from the telescope into the longitudinally-oriented external scientific instrument modules. This requirement would result in additional alignment problems and in additional losses due to light scattering that are not inherent with the design of Figure 3-12.

The concept of Figure 3-12 also provides a more efficient thermal design, since the internal modules are arranged in a configuration that more closely approximates a "conventional" satellite rather than an arrangement that exposes the mission equipment directly to the space environment. This results in its thermal provisions being more amenable to normal thermal analysis techniques and will result in lower weight penalties attributable to the thermal design.

If a hard-docked service module were used to effect the module change-out for the concept of Figure 3-9, it would be necessary that a rotatable structural ring approximately 2.74 m (9 ft) in diameter be provided in order to furnish the capability of module indexing and effect module extraction and replacement. Potential bearing design problems and problems associated with module indexing and replacement tolerances that may arise due to

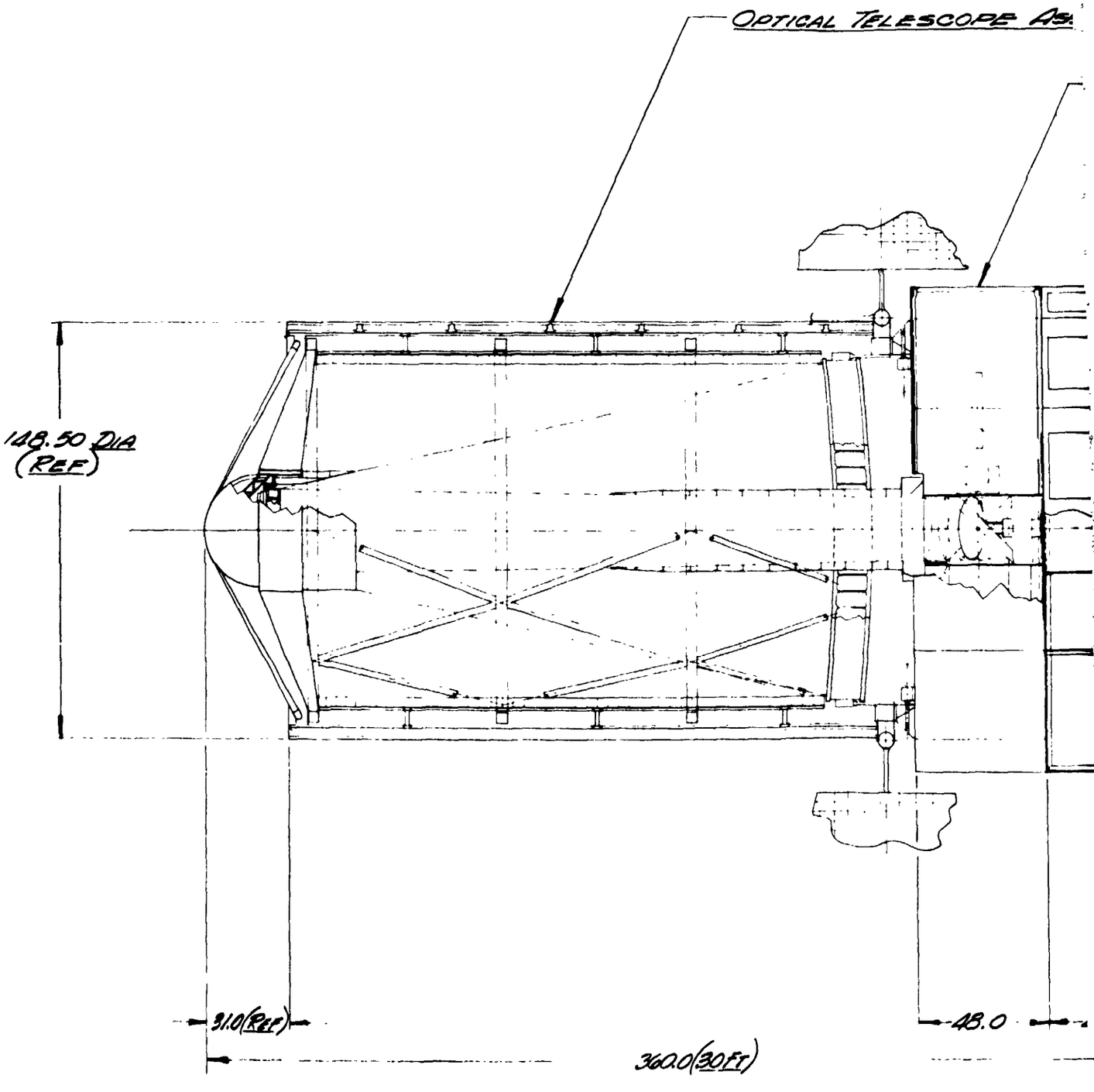
differential thermal loads have not yet been fully explored. Even though indexing would not be required if this configuration were used for module replacement by the manipulator, the other potential disadvantages noted above (light sealing, contamination, thermal, additional fold mirrors, necessity for revision of the Orbiter docking provisions, etc.) would still apply. Finally, the design of Figure 3-12 is superior to that of Figure 3-9 from the viewpoint of space utilization, since the internal volume provided by the latter concept is essentially wasted. The concept of Figure 3-12 provides approximately 30 percent more volume for mission equipment per unit length of module than that of Figure 3-9.

Consideration of the various geometric and subsystems aspects involved in the comparison of the subject concepts therefore indicates that the concept of Figure 3-12 exhibits most, if not all, of the advantages for use with the Orbiter-based manipulator. While the design of Figure 3-9 can be, as explained, amenable to use with either the manipulator or a hard-docked service module, it does not result in a "preferred" design when considered for use primarily with the Orbiter-based manipulator. On the other hand, the concept of Figure 3-9 is well adapted to module replacement of high-altitude satellites by means of a servicing vehicle in circumstances where the manipulator is unavailable.

b. Payload Module Geometry

Module replacement of low-earth-orbit maintainable payloads may be effected by the Orbiter manipulator system. There will be a dimensional tolerance range on the exact location to which the manipulator can rapidly and efficaciously move a module due to the inherent limitations of the (human) operator, the fits and clearances at the various joints of the manipulator, and end effector deflections and tip-off. It is desirable therefore to provide all possible module design features that can accommodate these circumstances and simplify the function of module withdrawal and insertion.

A layout drawing (Figure 3-14) was made to assist in the identification of basic design features and equipment that should characterize a typical



FOLDOUT FRAME

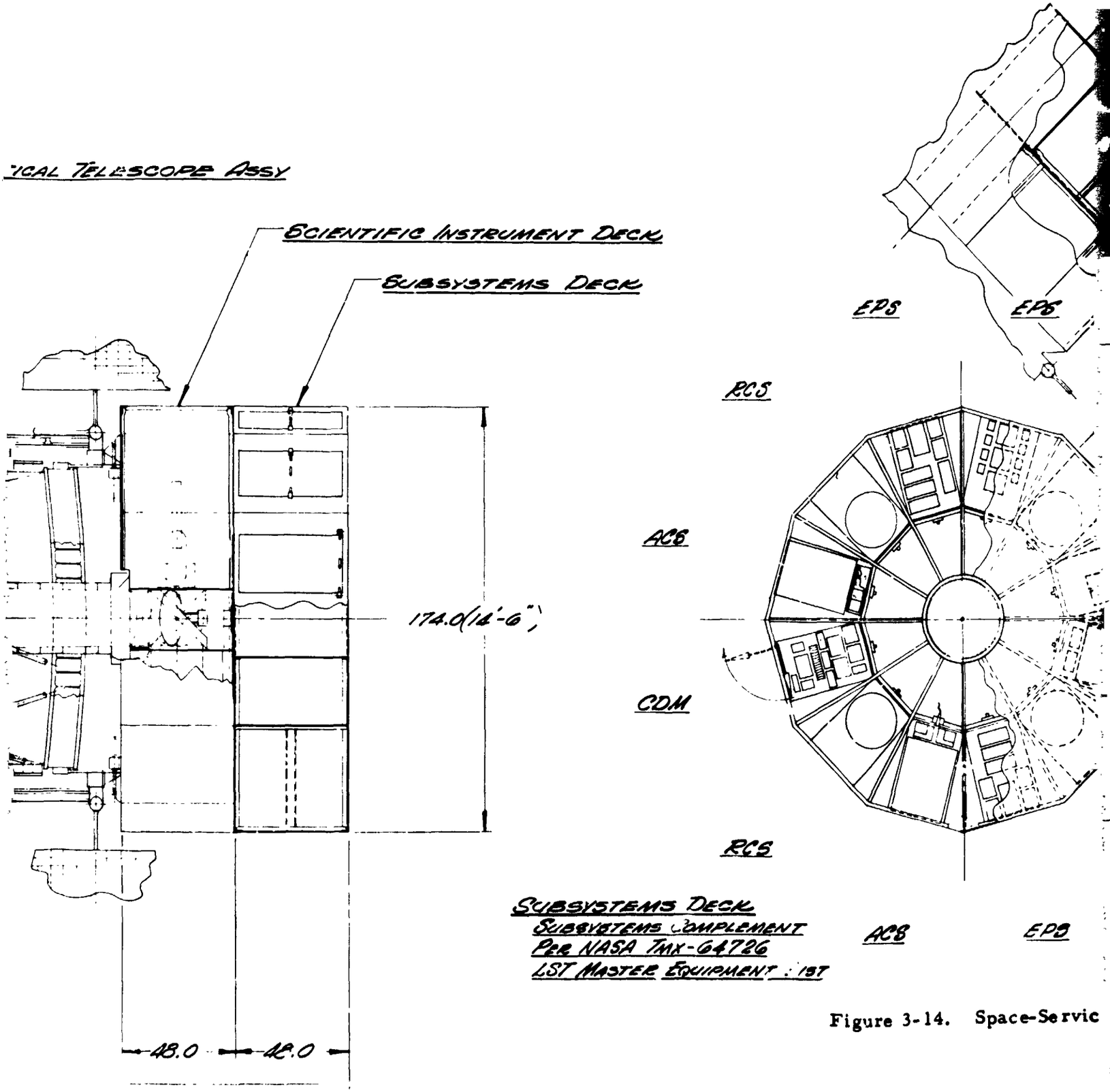


Figure 3-14. Space-Servic

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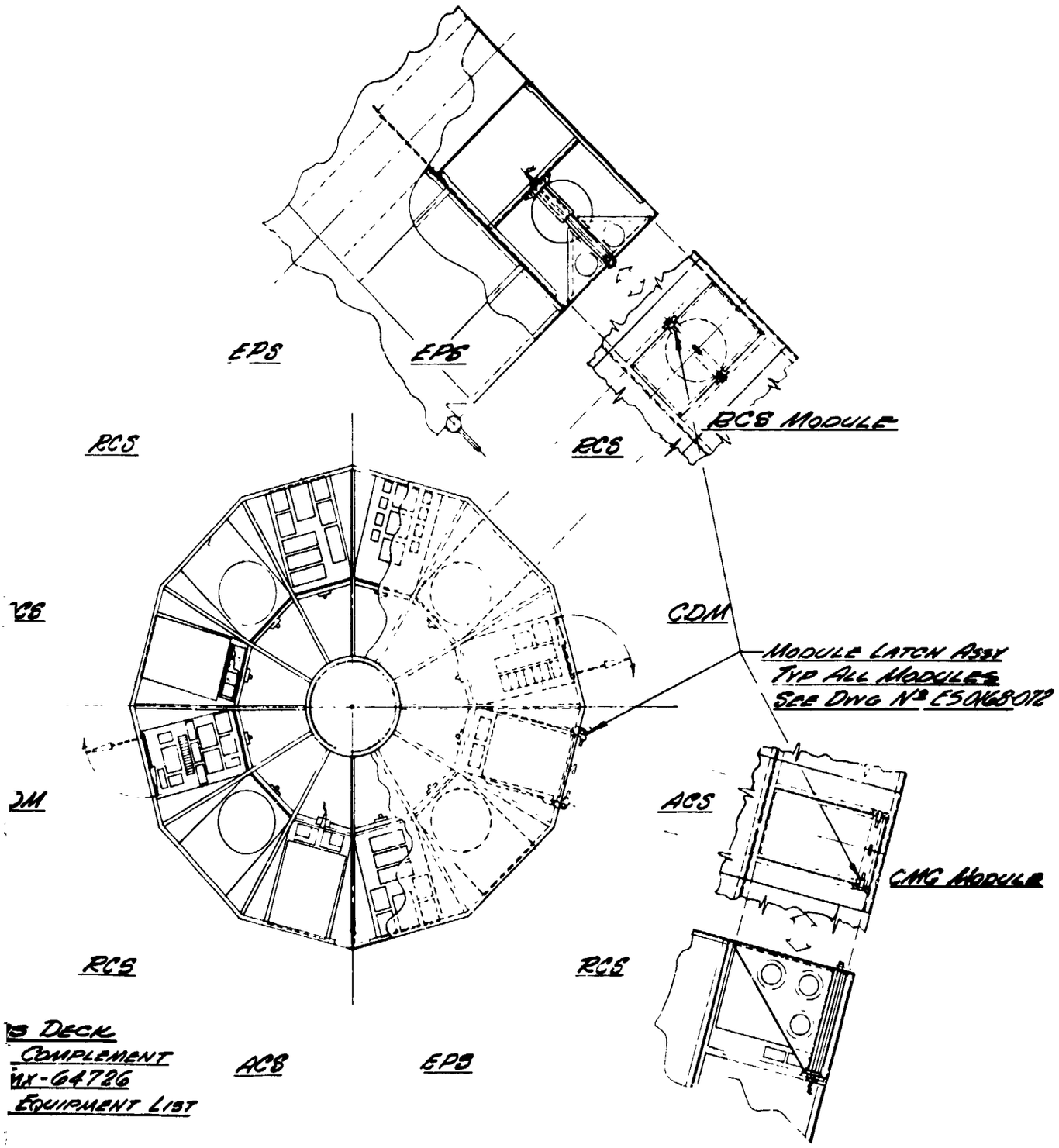


Figure 3-14. Space-Serviceable LST Spacecraft Corcept

FOLDOUT

module designed for replacement by the manipulator. Consideration of module/structure clearance provisions indicated by the manipulator handling requirements suggests a design that provides the basic characteristics and generous engagement tolerances of a probe and drogue.

This characteristic can be suitably approximated by a module base or equipment mounting plate whose sides are machined to a slight taper. A module baseplate with sides tapering only one or two degrees will (depending on the module length) provide a minimum clearance of several inches between the leading edge of a module and the spacecraft structure when a module is in the process of being inserted into the structure. This feature can be expected to accommodate module handling tolerances and ease the task of module replacement. Sides of the module that would ultimately be in sliding contact with structure can be provided with rub strips of teflon or other suitable material or coated. External protuberances (for example, rollers) that could be damaged by inadvertent or premature contact with structure should be avoided.

After the module has been partially inserted into the structure, a tapered pin centrally located in the leading (forward) face of the module can be caused to engage a tapered hole in the payload. The pin would also serve to provide a power and ground connection for the equipment located on the mounting plate. The central location of the pin will serve to protect it from handling damage after the leading edge of the module is initially inserted into the payload. A chamfer or "countersunk" surface can be provided at the entrance of the (pin) receptacle so that the pin and receptacle will always become engaged, even if the module is off to one side of the (receiving) structure during insertion. Suitable dimensioning (reduction in or removal of taper) at the trailing edge of the module can make the module essentially self-guiding in this area and provide a close fitting "seat" between the module and spaceframe when the module is fully inserted.

3. MASS PROPERTIES

The LST mass properties were primarily based on part counts from the SYSOPT program. The SYSOPT provided part counts to determine weight

growth with MMD for the subsystems that are represented in the reliability model. These consisted of the stabilization and control; communication, data processing, and instrumentation; and electrical power subsystems. The structures and environmental control subsystems were not included in the reliability model.

Since the structure was not included in the SYSOPT program, it was estimated from design studies. The expendable payload structure weight was based on Reference 5 (structure weight = 0.118 x equipment weight). The ground-refurbishable structural weight was estimated by increasing the structural weight for accessibility by factoring the expendable structure by 1.10. The structure weight for the on-orbit maintainable payload was based on design studies which are shown in Table 3-6. The payload consists of a basic spaceframe with 24 slots to accommodate space-replaceable modules; however, only 22 modules were required to package the parts.

The total LST weights for the expendable, ground-refurbishable, and on-orbit maintainable payloads are shown in Tables 3-7, 3-8, and 3-9, respectively. The reference weight is the man-tended version that is described in Reference 5.

The spacecraft synthesis program was not used, because the LST is a special type of payload that is not represented by the weight equation. The LST mission equipment weight is over 50 percent of the total weight, whereas the satellites used in the weight equations were generally around 15 to 20 percent of the total satellite weight.

4. MODULE SIMULATION PROGRAM

The parts for the three MMD configurations that were described in the Reliability Model section were allocated to 22 of 24 available modules provided by conceptual design. The parts were ordered by subsystems, assemblies, or components so that each module will contain equipments based on function criteria. The size and weights were the limiting factors in the decision-making process in the amount of parts in each module. Using this approach, the parts were assigned to the modules as shown in Table 3-10.

Table 3-6. LST Structural Weights for On-Orbit Maintainable Payload

	MMD					
	0.5 yr		1.0 yr		1.5 yr	
	kg	lb	kg	lb	kg	lb
Basic Structure	182	400	185	406	192	422
Module Tracks	262	576	262	576	262	576
Base Plates	391	860	391	860	391	860
Mechanism	136	300	136	300	136	300
Equipment Supports	68	150	68	150	68	150
Miscellaneous	386	850	388	854	392	864
Total Structure	1,422	3,136	1,427	3,146	1,439	3,172

Table 3-7. Shuttle-Launched LST Weight, Expendable Payload

ITEM	REFERENCE		EXPENDABLE					
			MMD (YEAR)					
	1.25		0.5		1.0		1.5	
	kg	lb	kg	lb	kg	lb	kg	lb
Structure	834	1834	809	1780	817	1800	845	1860
TPS	72	159	72	159	72	159	72	159
SSM Contamination	29	64	29	64	29	64	29	64
Stabilization & Cont.	619	1363	524	1154	575	1267	653	1437
CDPI	71	157	63	139	68	149	101	221
Electrical	686	1510	577	1270	610	1343	716	1577
Mission Equipment	5580	12307	5580	12307	5580	12307	5580	12307
Dry Weight	7891	17394	7654	16873	7751	17089	7996	17625
ACS Propellant	20	43	21	46	42	92	78	172
Gross Weight	7911	17437	7674	16919	7793	17181	8074	17797
Docking Equipment	-	-	-	-	-	-	-	-
Adapter (6%)	475	1046	461	1015	469	1031	485	1068
Launch Weight	8386	18483	8135	17934	8262	18212	8559	18865

Table 3-8. Shuttle-Launched LST Weight, Ground-Refurbishable Payload

ITEM	REFERENCE		GROUND REFURBISHMENT					
			MMD (YEAR)					
	1.25		0.5		1.0		1.5	
	kg	lb	kg	lb	kg	lb	kg	lb
Structure	834	1834	889	1958	900	1980	930	2046
TPS	72	159	72	159	72	159	72	159
SSM Contamination	29	64	29	64	29	64	29	64
Stabilization & Cont.	619	1363	524	1154	575	1267	653	1437
CDPI	71	157	63	139	68	149	101	221
Electrical	686	1510	577	1270	610	1343	716	1577
Mission Equipment	5580	12307	5580	12307	5580	12307	5580	12307
Dry Weight	7791	17394	7734	17051	7836	17269	8081	17811
ACS Propellant	20	43	21	46	42	92	78	172
Gross Weight	7911	17437	7754	17097	7877	17361	8159	17983
Docking Equipment	-	-	45	100	45	100	45	100
Adapter (6%)	475	1046	466	1026	474	1042	490	1079
Launch Weight	8386	18483	8265	18223	8397	18503	8694	19162

Table 3-9. Shuttle-Launched LST Weight, On-Orbit Maintainable Payload

ITEM	REFERENCE		ON-ORBIT MAINTAINABLE (24 Modules)					
			MMD (YEAR)					
	1.25		0.5		1.0		1.5	
	kg	lb	kg	lb	kg	lb	kg	lb
Structure	834	1834	1422	3136	1427	3146	1439	3172
TPS	72	159	72	159	72	159	72	159
SSM Contamination	29	64	29	64	29	64	29	64
Stabilization & Cont.	619	1363	524	1154	575	1267	653	1437
CDPI	71	157	63	139	68	149	101	221
Electrical	686	1510	749	1650*	783	1723*	888	1957*
Mission Equipment	5580	12307	5580	12307	5580	12307	5580	12307
Dry Weight	7891	17394	8439	18609	8534	18815	8762	19317
ACS Propellant	20	43	23	51	46	101	85	188
Gross Weight	7911	17437	8462	18660	8580	18916	8847	19505
Docking Equipment	-	-	45	100	45	100	45	100
Adapter (6%)	475	1046	509	1123	517	1141	533	1176
Launch Weight	8386	18483	9016	19883	9142	20157	9425	20781

*Includes 8.6 kg (19 lb) x 20 modules for power distribution.

Table 3-10. LST Space-Replaceable Module Characteristics for a 24-Module Concept

SRM No.	Part No.	Part Title	Weibull Function								
			0.5 yr.			1.0 yr.			1.5 yr.		
			α	β	MMD	α	β	MMD	α	β	MMD
1	30 31	Gyro and Electrical Fixed Head Star Track	45.96	1.321	42	87.36	2.234	74.52	111	3.1	93
2, 5, 8, 11	62	Thrusters and Expendables	18.85	9.45	18	31.15	14.15	30	53.66	15.75	52
3, 9	42 43 44 45	Solar Power Distribution Deploy Mech Orient Drive Charger/Battery Regulator	33.6	1.457	30	45	1.75	40	64.5	2.75	57
4	46 47 48 49 50	Electrical Control Assembly Electrical Dist Unit (EDU) Electrical Dist Unit (EDU) Electrical Dist Unit (EDU) Electrical Dist Unit (EDU)	372.8	1.171	107	367.3	1.186	107	512.8	2.532	119
6, 7, 10	33	Control Moment Gyro	4166	1.036	119	4721	1	119	4679	1	119
12	32 34 38 39 40	Transfer Assembly Digital Processor Assembly Magnetic Torquer Electric Magnetic Torquer Electric Magnetic Torquer Electric	74.48	1.43	62	97	2.3	81	236.4	2.182	112
13	None										
14	4	F/96 Cameras A, B, C	117.1	1	75	117.1	1	75	117.1	1	75
15	14	F/12 Camera Assembly	930.9	1	113	930.9	1	113	930.9	1	113
16	8	Hi Reso. Specto 204-333 km (110-180 nmi)	1225	1	115	1225	1	115	1227	1	115

Table 3-10. LST Space-Replaceable Module Characterist.cs for a 24-Module Concept (Continued)

SRM No.	Part No.	Part Title	Weibull Function								
			0.5 yr.			1.0 yr.			1.5 yr.		
			α	β	MMD	α	β	MMD	α	β	MMD
17	9	Hi Reso Specto 333-648 kg (180-350 nmi)	1225	1	115	1225	1	115	1225	1	115
18, 24	51	[illegible]	280.1	1.057	99	387.7	1.26	109	509.4	1.949	118
	52	[illegible] onders									
	53	Phase Shift Keyed Demod									
	54	Freq. Mux									
	55	Remote Comm. Decoders									
	56	Remote Comm Decoders									
	57	Remote Comm Decoders									
	58	Remote Comm Decoders									
	59	Remote Comm Decoders									
	60	Remote Comm Decoders									
	61	Format Gen. PCM Encoder									
19	None										
20	10	Faint Obj. Specto 204-407 km (110-220 nmi)	278.7	1	98	278.7	1	98	278.7	1	98
21	11	Faint Obj Specto 407-1222 km (220-660 nmi)	438.5	1	105	438.5	1	105	438.5	1	105
22	12	Faint Obj Specto 1222-1852 km (660-1000 nmi)	1037	1	113	1037	1	113	1037	1	113
23	13	Faint Obj Specto 1852-9260 km (1000-5000 nmi)	7493	1	119	7493	1	119	7493	1	119
GRM	1	Field Select Mirror Assembly	105.9	1.333	77	105.9	1.333	77	105.9	1.333	77
	2	Camera Shutter Flag									
	3	Camera Select Mechanism									
	5	Slit Jaw Camera									
	6	Slit Mechanism									
	15-19	Optical Telescope Assembly									
	35-37	Magnetic Torquer									
	41	Solar Panels									

The space-replaceable modules number is schematically shown in Figure 3-15. The aft deck has modules 1 through 12, and the forward deck has modules 13 through 24. Module numbers 13 and 19 are without parts in this example case but are available to accommodate future modifications.

With the parts assigned to the modules, the reliability characteristics of each module were computed. The Weibull functions were then determined for each module and are shown in Table 3-10 for the three MMD configurations. The MMD of each module is also included to indicate the modules with low reliability which are the thrusters and solar power in the housekeeping subsystems and in a pseudo fashion, the expendables.

Since the Weibull function is a best-fit of the predicted module reliability, it is of interest to examine the comparison. This comparison is made in Figure 3-16 for space-replaceable module No. 1. It indicates that the fit is relatively close to the module predicted failure. At the system level, when all of the modules are included, the fit is better. This can be seen in Figure 3-17, where the one-year MMD configuration is a near duplication of the predicted reliability; however, a departure is noted beyond the design MMD.

With the Weibull functions determined for each module, the expected numbers were then determined using the SAMS program. The result of this computation is shown in Figure 3-18 for the three MMDs and repair strategies. The number of modules replaced per visit can be observed to have a strong influence on the expected number of launches per year. By replacing more than the failed module, the expected number reduces substantially for conditional and perfect-failure prediction cases and reaches an apparent maximum effect at about 30 percent module replacement. These expected numbers were supplied to the payload program cost program.

5. PAYLOAD PROGRAM COST

The LST costs were determined as described in Payload Program Cost Estimating. These were computed using the remote terminal for various MMDs, program durations, repair cost factors, trip charges, and repair

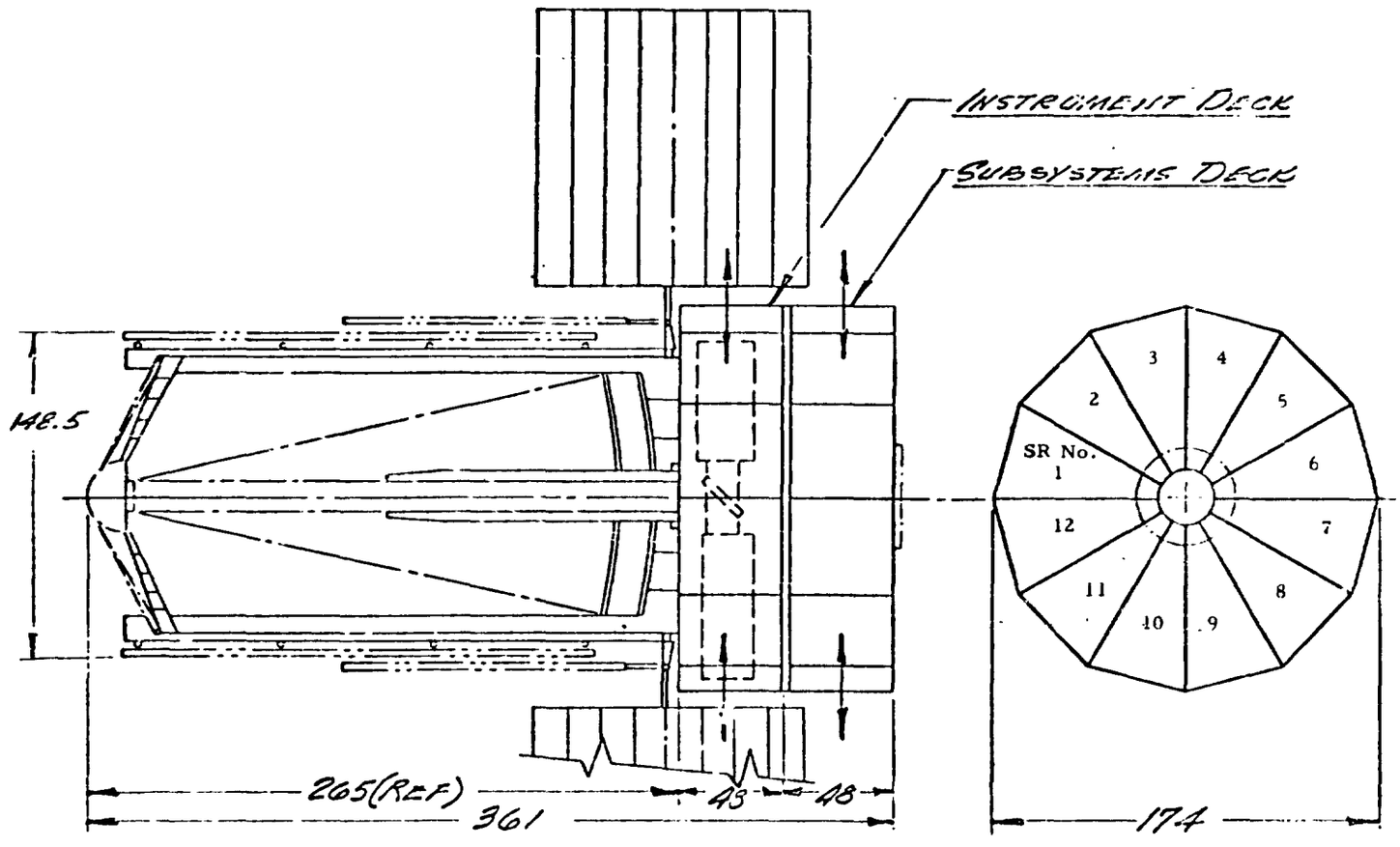


Figure 3-15. Modularized LST, Concept No. 5

MMD	SRM #1 Parts List		Weight	
	Gyro and Electronics	Fixed Head Star Tracker	kg	lb
B/L	6	3	32.2	71
0.5	4	2	21.4	47
1.0	6	3	32.2	71
1.5	8	4	42.6	94

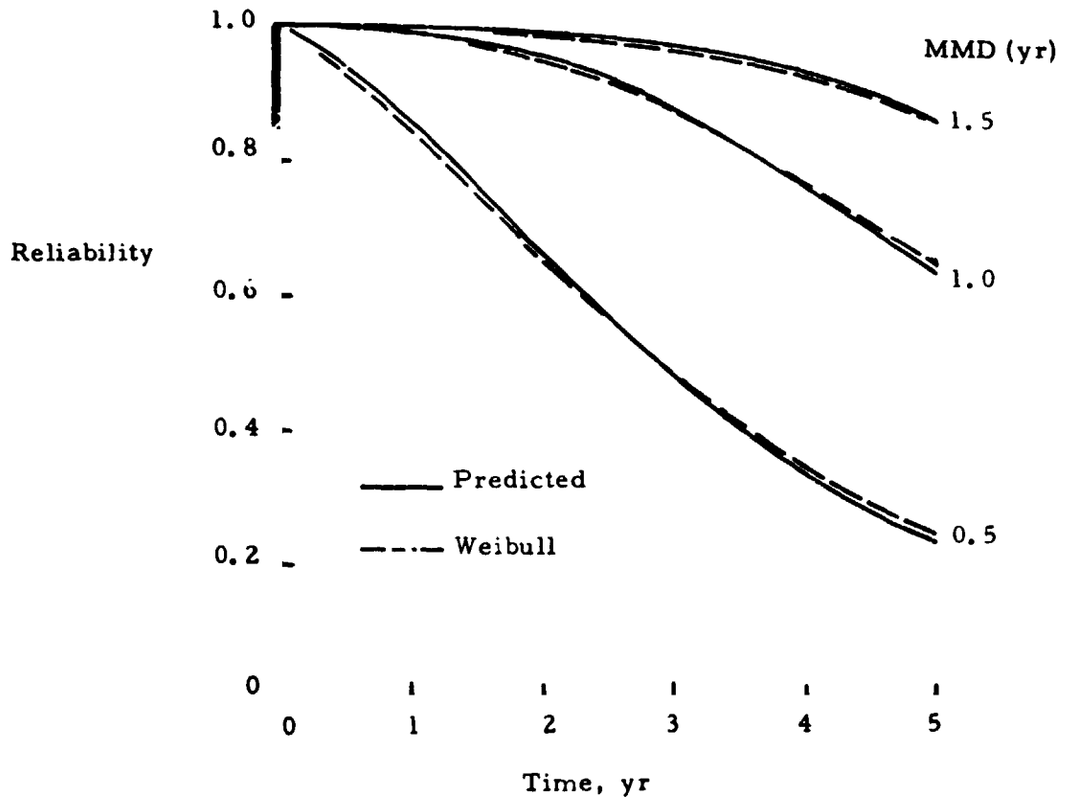


Figure 3-16. Reliability Characteristics for LST Space-Replaceable Module No. 1

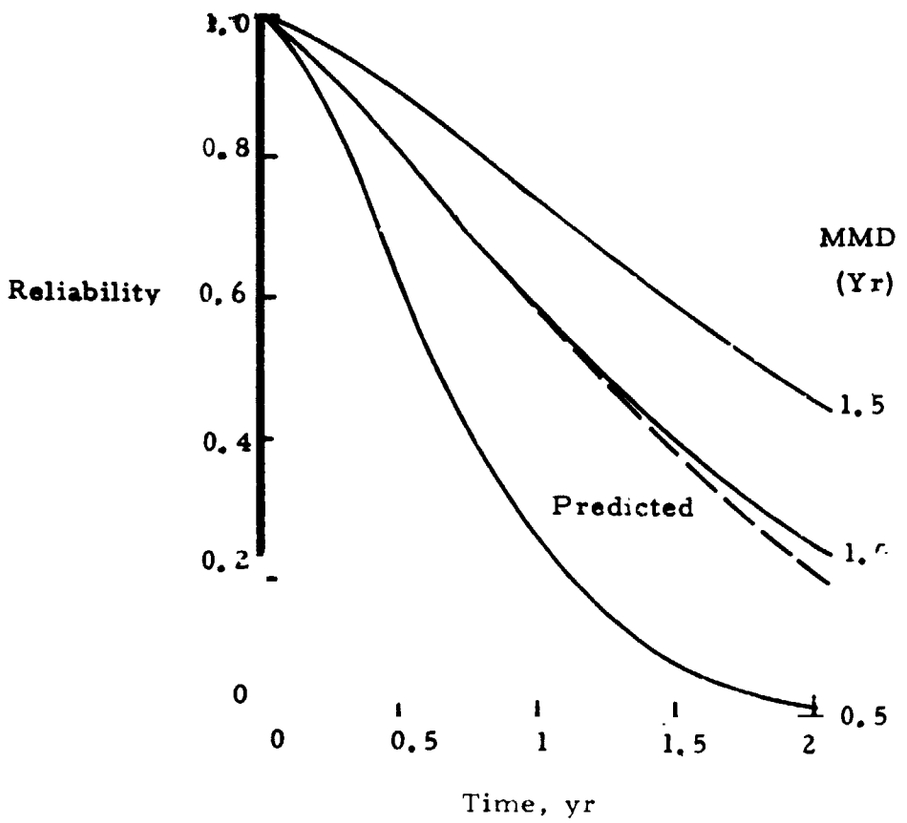


Figure 3-17. LST System Reliability Based on Weibull Function

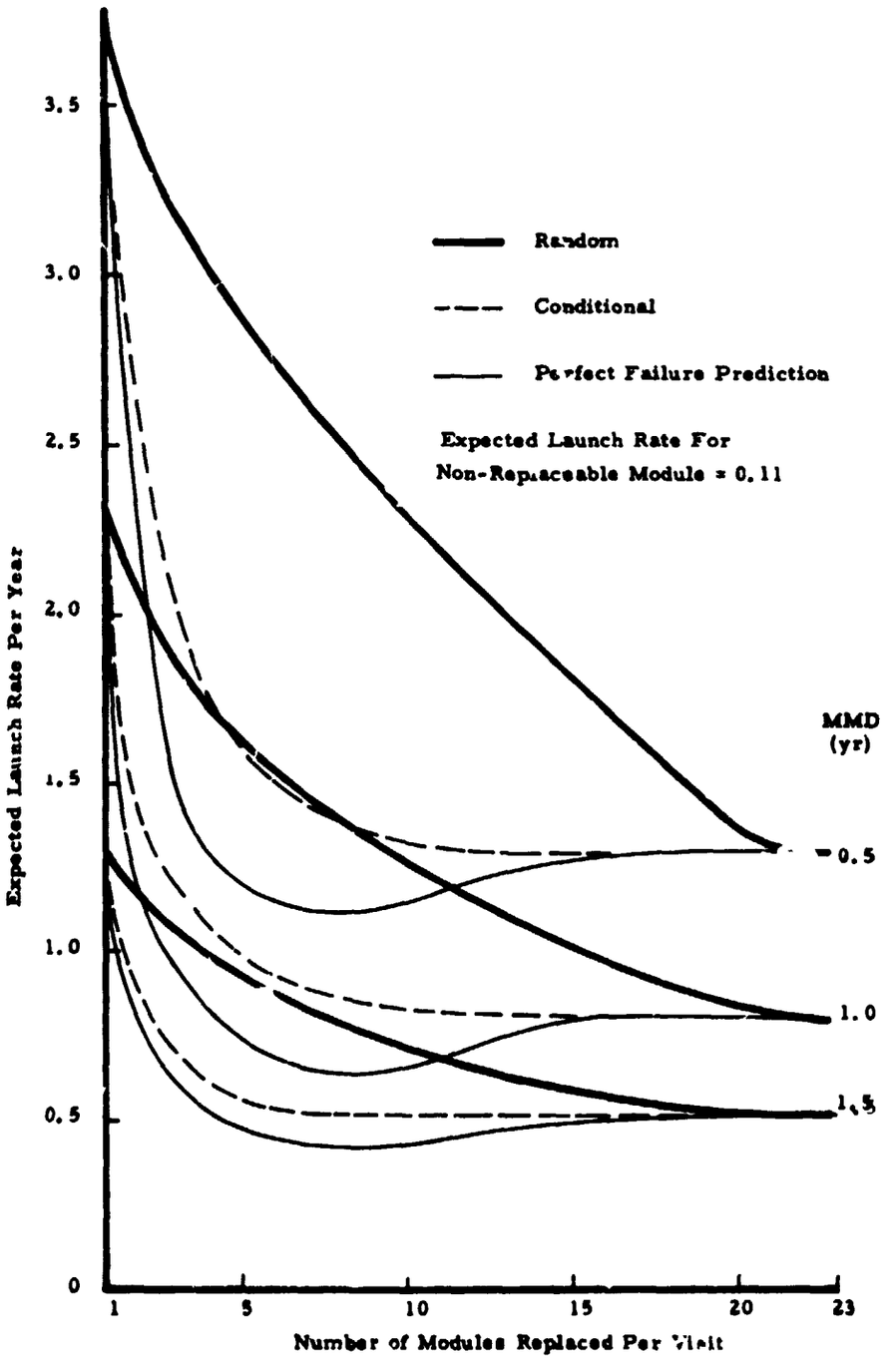


Figure 3-18. Expected Launch Rate for LST

strategies. These costs were based on a single satellite on-orbit and are not applicable for a multi-satellite on-orbit system.

The payload unit cost and total program cost are shown in Figure 3-19 as related to MMD for the three types of payloads (expendable, ground-refurbishable, and on-orbit maintainable). The unit cost is the payload recurring cost, and the total program cost includes unit cost, RDT&E, and operations costs. The unit cost increases as the MMD increases due to the increase in parts redundancy. The higher unit cost of an on-orbit maintainable payload over the expendable and ground-refurbished types reflects the effect of modularity. The total program cost decreases as the MMD increases for the expendable payloads but remains relatively constant for the on-orbit maintainable payloads. The expendable program cost drops because of the reduction, mainly in the number of payload buys but also in the expected number of launches The on-orbit maintainable program cost showed a lower, fairly constant total cost, because the number of payload buys is the same over the range of MMD considered. The total program cost of the ground-refurbishable payload was between the on-orbit maintainable and expendable payloads. The total program cost shows that in a short-duration program the number of payload buys is the primary cost driver. The 1.5 year program duration basically equalizes the new buys among the three types of payloads and results in nearly equal total program cost for the three types of payloads.

In Figure 3-20, the total program cost is shown as the program duration is varied. The on-orbit maintainable payload has a lower cost increase than the expendable payload The expendable payloads show a cost advantage for short-duration programs while the on-orbit maintainable payload shows lower costs as the program is extended. These costs assumed only one RDT&E effort over the program duration which would indicate that the comparison is realistic only over that program life which does not require a major new development effort.

To examine the effect of repair cost on the total program cost, the on-orbit maintenance repair cost factor and the ground-refurbishment repair cost factor were varied about their nominal values of 0.25 and 0.39,

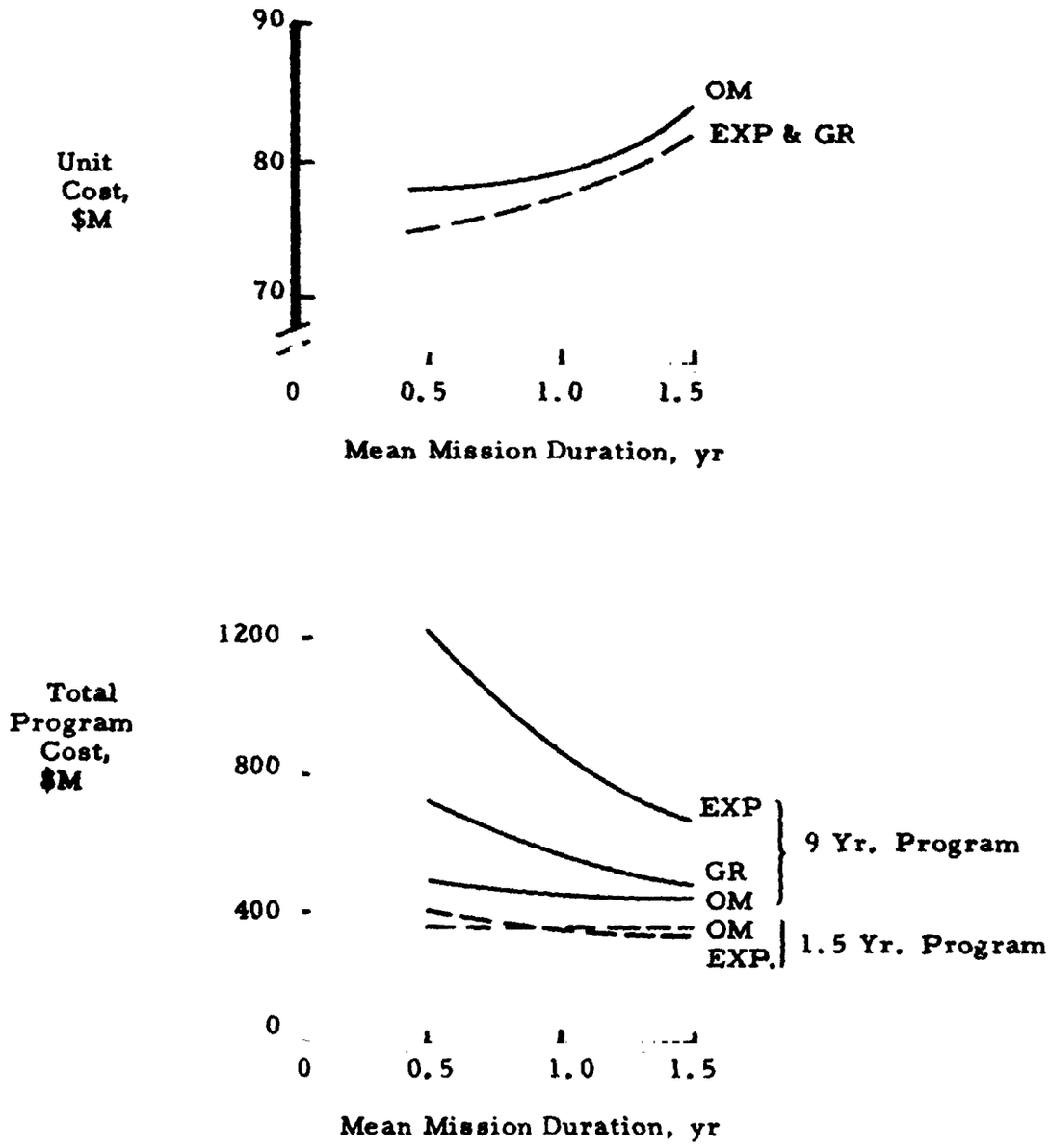


Figure 3-19. Effect of Mean Mission Duration on LST Unit and Program Cost

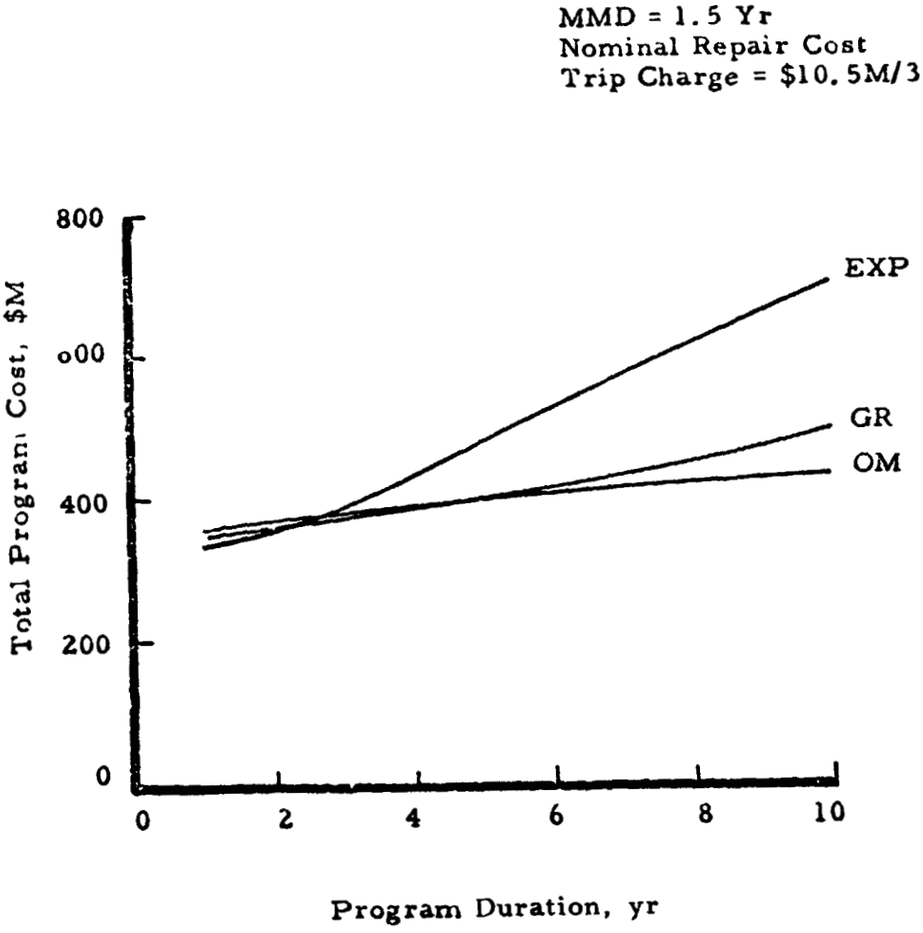


Figure 3-20. Effect of Program Duration on Total Program Cost for LST

respectively. This result is shown in Figure 3-21 where the on-orbit maintainable cost remains below the ground-refurbishable cost over the cost factor range and where both service concepts remained below the expendable payload. The difference in the total payload cost for the two service mode concepts is primarily due to the repair cost. The ground-refurbishment concept assumes that the complete payload will be refurbished, whereas the on-orbit maintainable concept assumes that refurbishment is only performed on the modules that are retrieved and returned. The other differences resulting from higher unit cost, higher expected number of launches, and lower launch-operation costs for on-orbit maintainable over ground-refurbishment payloads are small in comparison to the difference in service cost.

The trip charges were varied from dedicated launches to sharing the trip charges among five separate missions. The total program cost, as a function of trip charge, is shown in Figure 3-22 for the three types of payloads. It can be observed that this effect is small for a single satellite on-orbit. The cost slope is low and the cost ranking did not change.

One aspect of on-orbit maintenance affecting a low-cost approach for long-duration missions is the repair strategy. The on-orbit strategy is a function of the number of modules to be replaced per visit and the failure prediction method. The effects of these two factors on the total program cost is shown in Figure 3-23. The cost reduces substantially if more than the failed module is replaced, up to an optimum at about 30 percent of all the space-replaceable modules in the payload. The conditional failure prediction is based on most probable failure times, and the perfect-failure prediction (PFP) is based on accurate failure prediction; i.e., there is no time deviation about the failure time. To achieve PFP will require extensive instrumentation, telemetry, and data analysis; however, with nominal module instrumentation, one would expect that failed modules, penultimate failed modules, degrading modules, and modules with depleted consumables can be diagnosed accurately. Thus, the identification of modules to be replaced should be better than conditional and should approach PFP capability. The cost reduction is not significant enough to expend additional RDT&E funds

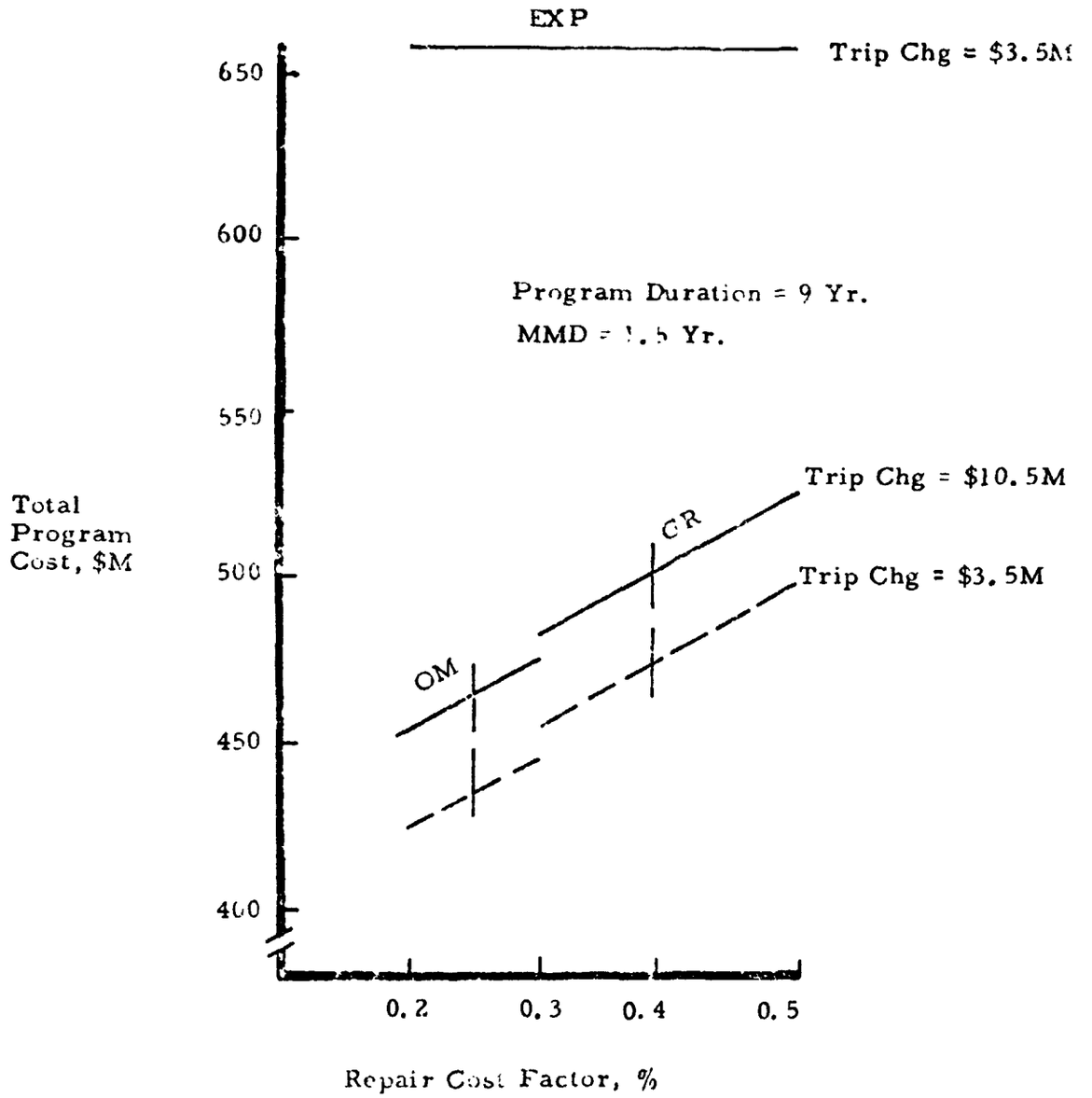


Figure 3-21. Effect of Repair Cost Factor on Total Program Cost for LST

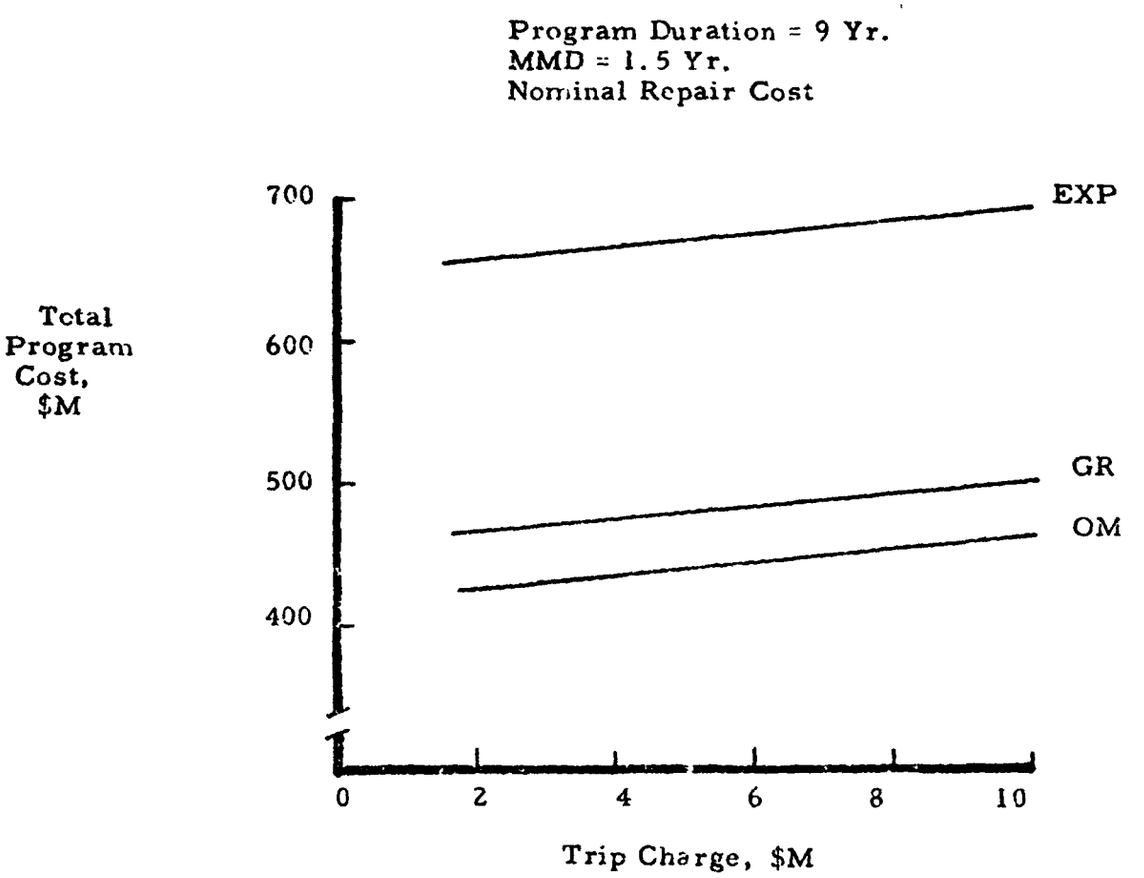


Figure 3-22. Effect of Trip Charge on Program Cost for LST

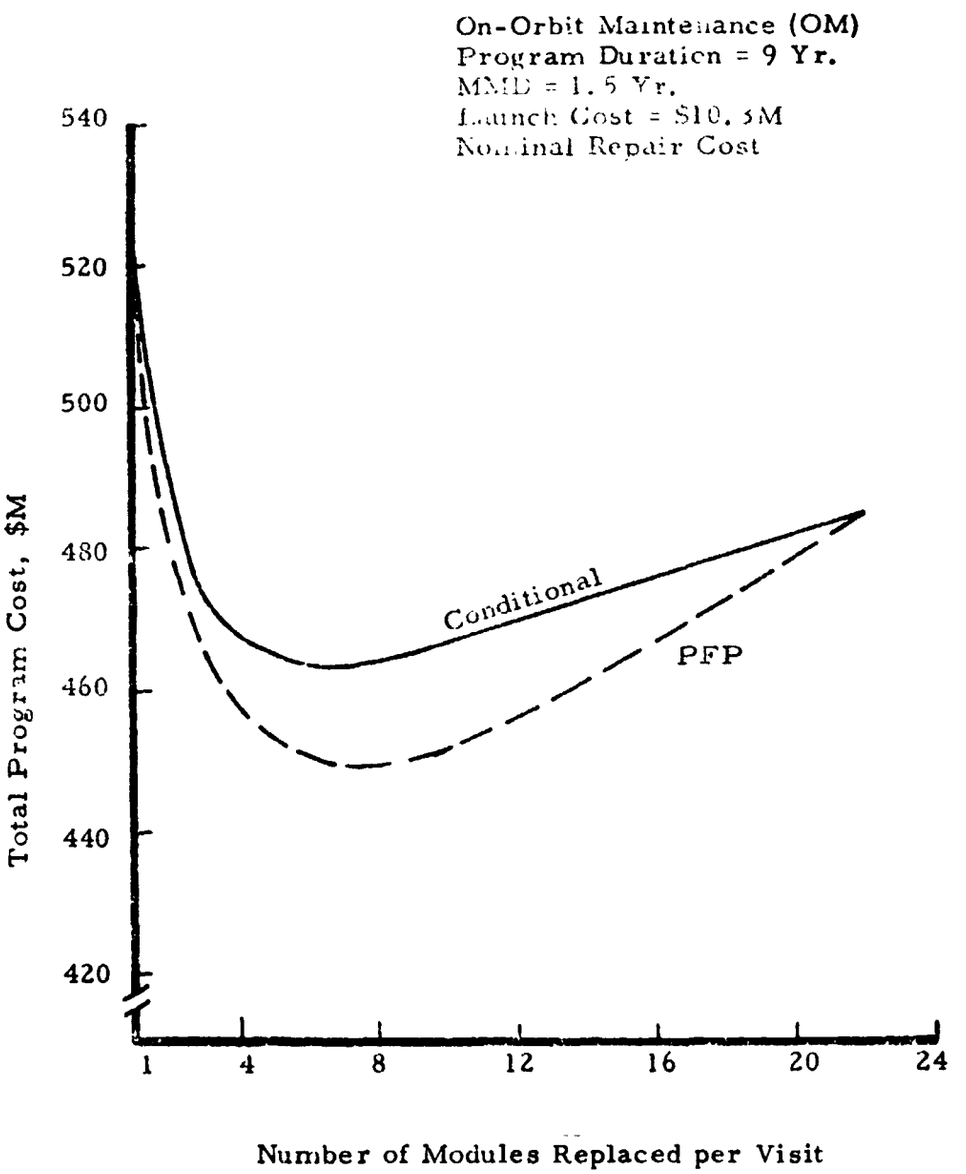


Figure 3-23. Optimum Number of Modules Replaced for LST with 22 Space-Replaceable Modules

to achieve PFP beyond that for instrumentation required to identify module failure modes.

From the cost analysis of the LST single payload on-orbit, the type of payload has been shown to be the most significant parameter influencing the cost. The program duration appears to determine the type of payload resulting in a low-cost approach. The repair cost factor and trip charge did not have significant impact on program cost. The proper repair strategy for on-orbit maintainable payloads appears to be to replace 30 percent of the modules per visit.

C. SHUTTLE-LAUNCHED DEFENSE SUPPORT PROGRAM DESIGN ANALYSIS

The Defense Support Program (DSP) satellite was selected for analysis, because it represents an applications-type payload in a high-altitude orbit requiring a Tug. Also, design information and costs on this satellite were documented and complete enough to permit the analysis to be pursued.

The current DSP is a spin-type payload, but it was modified in this study to a three-axis stabilized configuration. The payload configuration used in this study is not a recommended Shuttle-Launched DSP (SLDSP) design, but is a representative payload with adequate technical and cost data. The analysis reported here used the technical data from Reference 8 to configure the SLDSP. This analysis differs from previous studies in that it was performed to evaluate the effect of payload parameters on total program cost and not to recommend a design or to produce a point design. Therefore, the payload modularity and reliabilities were varied over a wide range in this analysis. Moreover, the number of satellites in the system was limited to one, and the effect of a multi-satellite system was not investigated.

1. RELIABILITY MODEL

The SLDSP reliability diagrams for the SYSOPT program were developed and the pertinent redundancy, failure rate, and weight information were compiled for the Shuttle-launched version of the DSP from available sources to minimize estimating. After the SLDSP was modeled, the

system was then reduced to a single-string configuration by removing the redundancies.

The reliability block diagrams were developed from References 8 and 9 and are shown in Figures 3-24 through 3-32 for the housekeeping portion only. Included with the reliability model are the failure rate, type of redundancy, and weight. The propulsion subsystem was changed to adapt it to the Shuttle. The attitude control subsystem was altered to a three-axis-stabilized type from a yaw-spin-type stabilization concept. In the areas where the level of reliability data needed was not provided in the referenced studies, previous contractor reliability analyses on the satellite were used. These include failure rates, duty cycles, internal redundancies, and modes of operation.

The assumptions made in this task were as follows:

a. The primary and backup units were identical.

b. Expendables for one-year operation were used in the single-string configuration. The added expendables to the propulsion subsystems for three-month blocks were 7.25 kg (16 lb) expendables and 2.92 kg (6.4 lb) hardware.

c. The units that were not made redundant were earth sensors (RADEC), adapters, inertial balancing mechanism, switches, and spaceframe.

d. Components do not approach wearout within 10 years.

e. Standby failure rates were 10 percent of active failure rates.

f. Equivalent failure rates were estimated for boxes without failure data by estimating the reliability at 15 months to compute the constant failure rate.

The reliability data were inputted into the SYSOPT and the optimum redundancies based on weight were obtained for various payload MMDs. This characteristic is shown in Figure 3-33 starting from a single-string MMD of about 0.5 years to a maximum MMD of about 4.8 years. For this study, MMDs of 1, 2, and 4.5 years were selected for detailed analysis.

T' parts count for each unit modeled in the reliability diagram (except for the mission equipment) for SLDSPs with the three MMDs is listed in Table 3-11. This part count was then used for the conceptual design

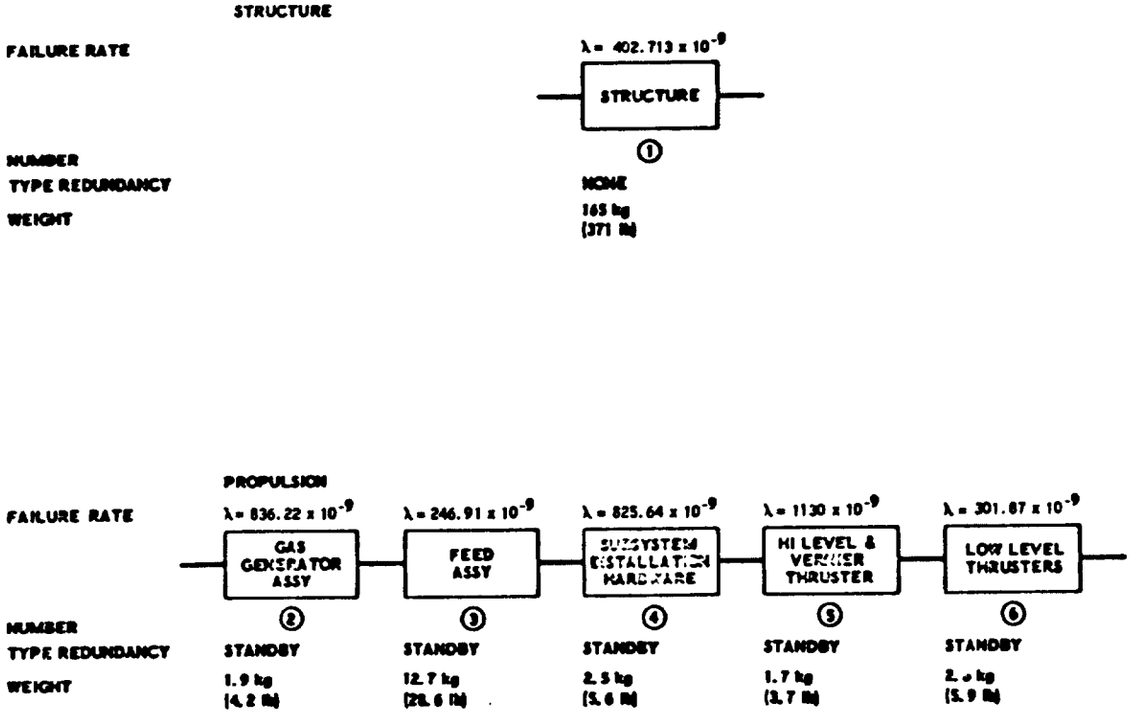


Figure 3-24. SLDSP Structures and Propulsion Reliability Model

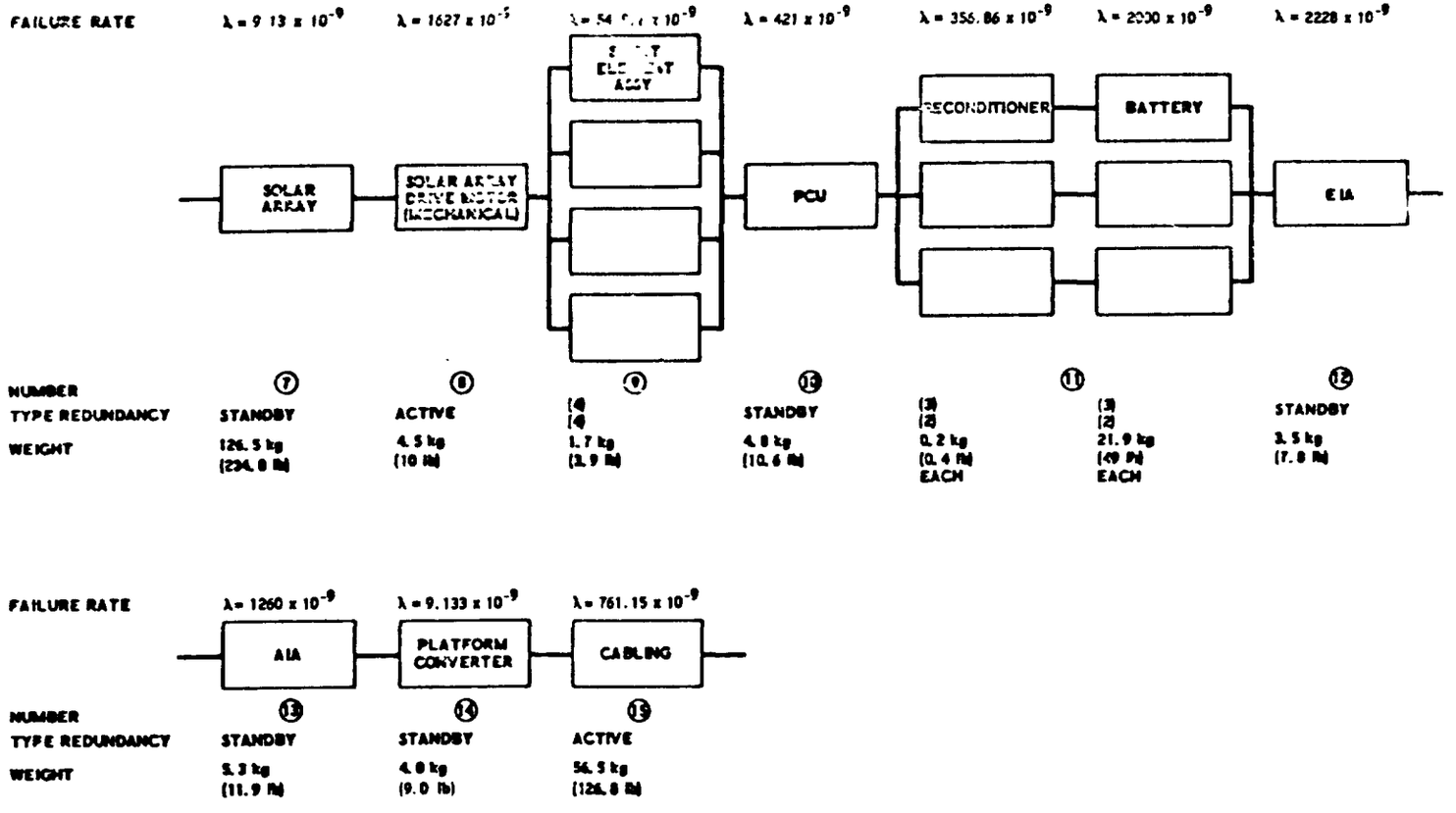


Figure 3-25. SLDSP Electrical Power Reliability Model

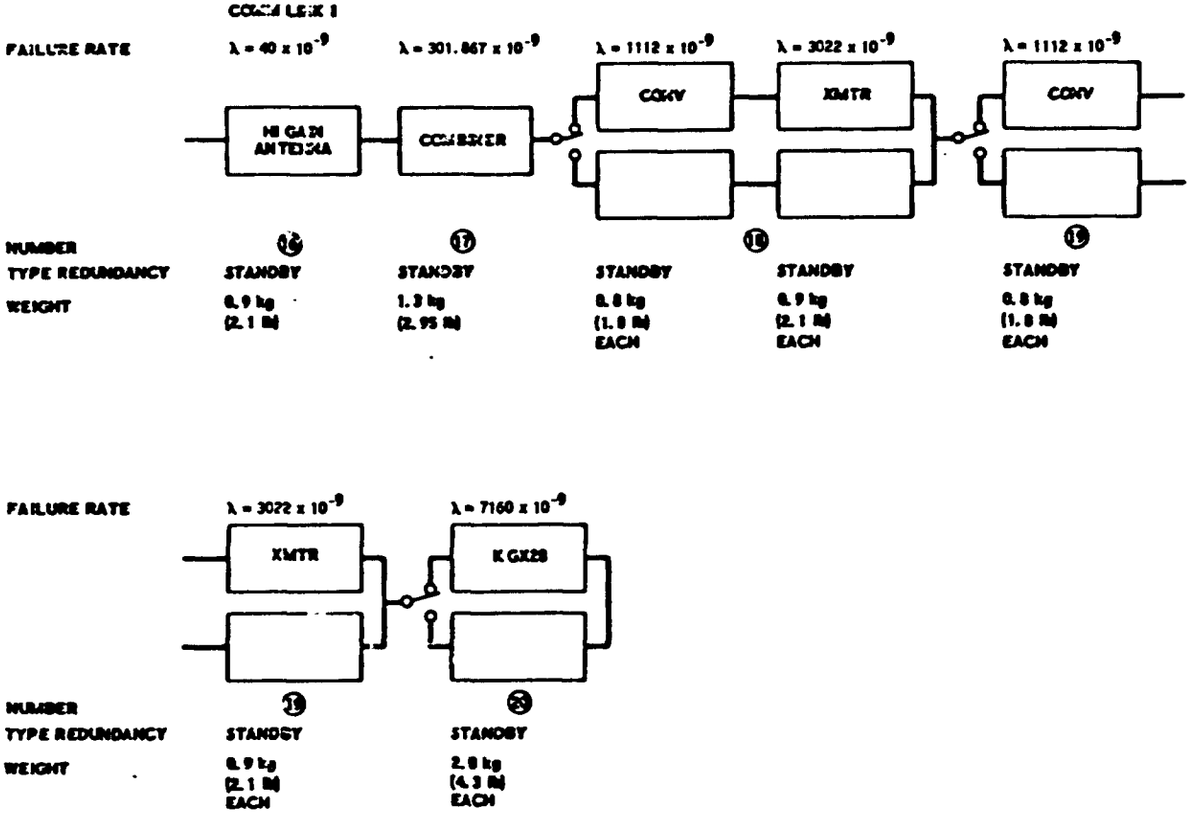


Figure 3-26. SLDSP Communication Link 1 Reliability Model

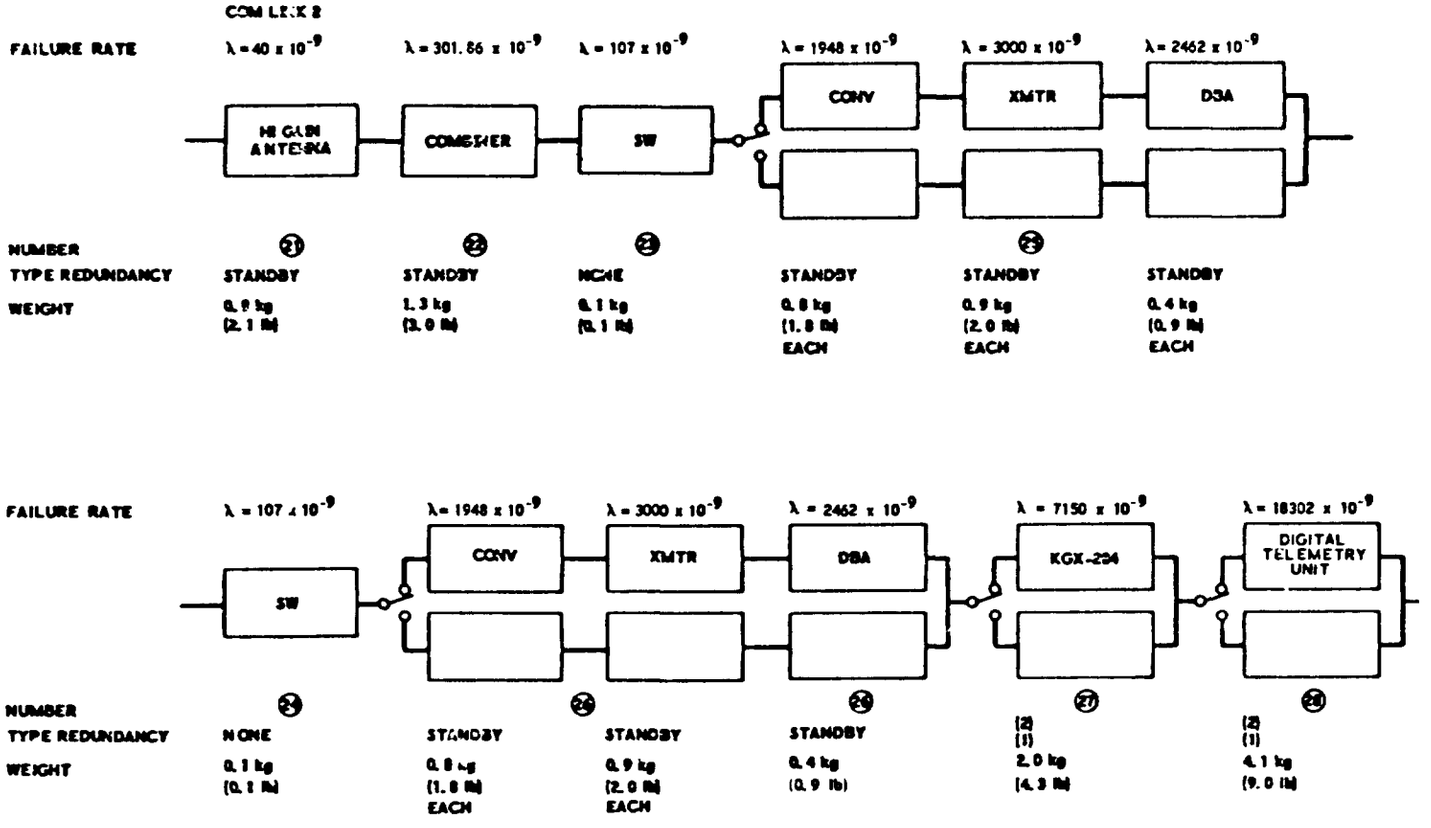


Figure 3-27. SLDSP Communication Link 2 Reliability Model

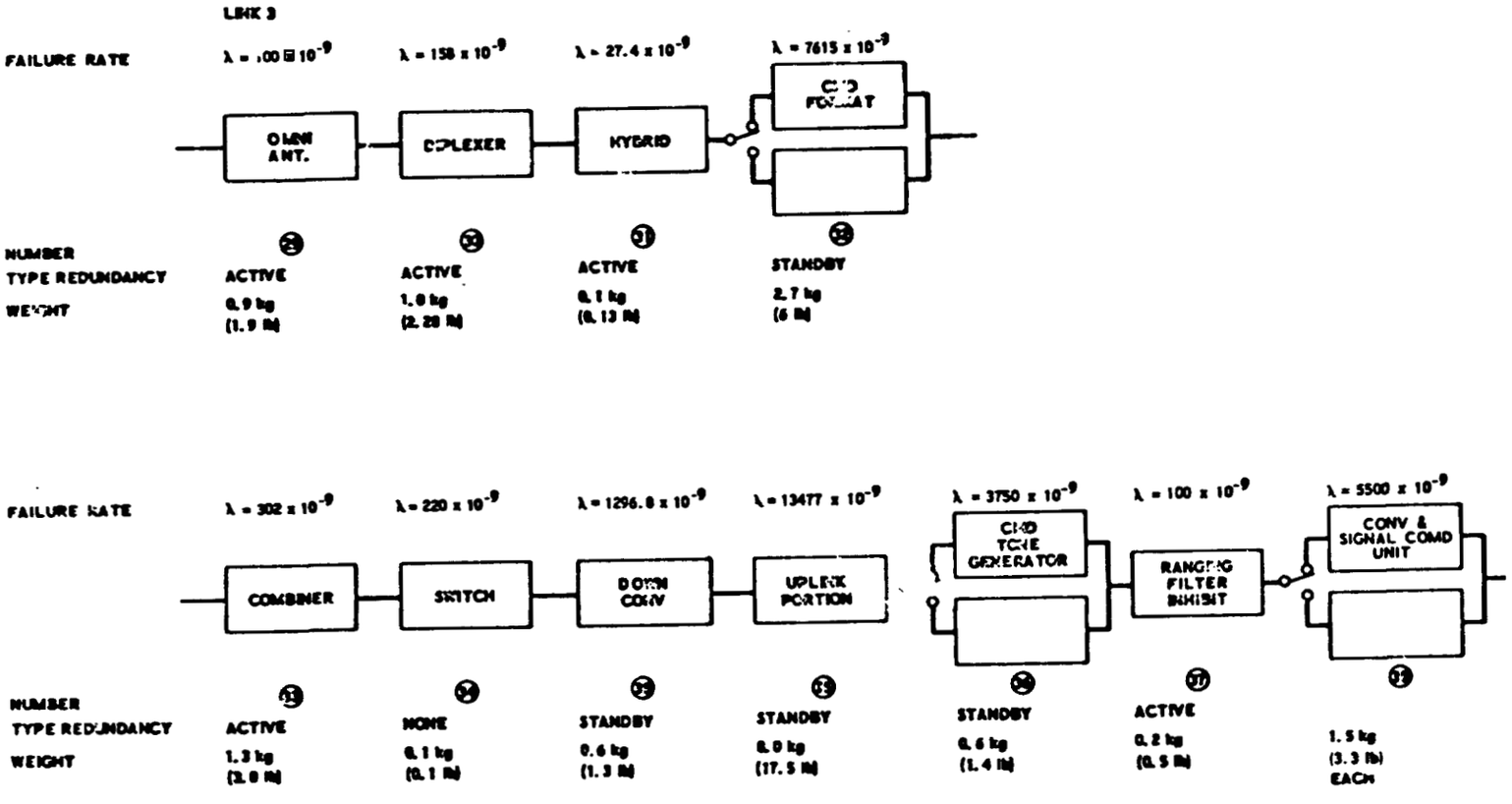


Figure 3-28. SLDSP Communication Link 3 Reliability Model

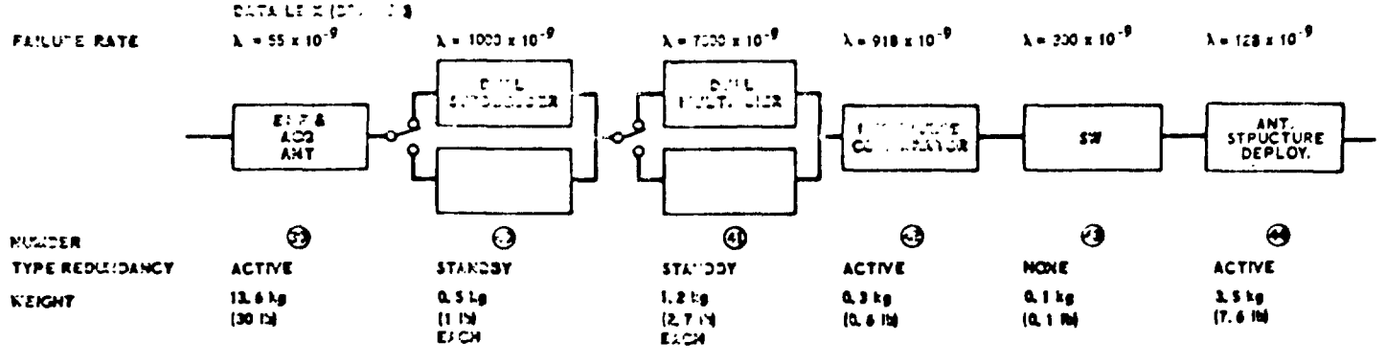


Figure 3-29. SLDSP Communication Data Link Reliability Model

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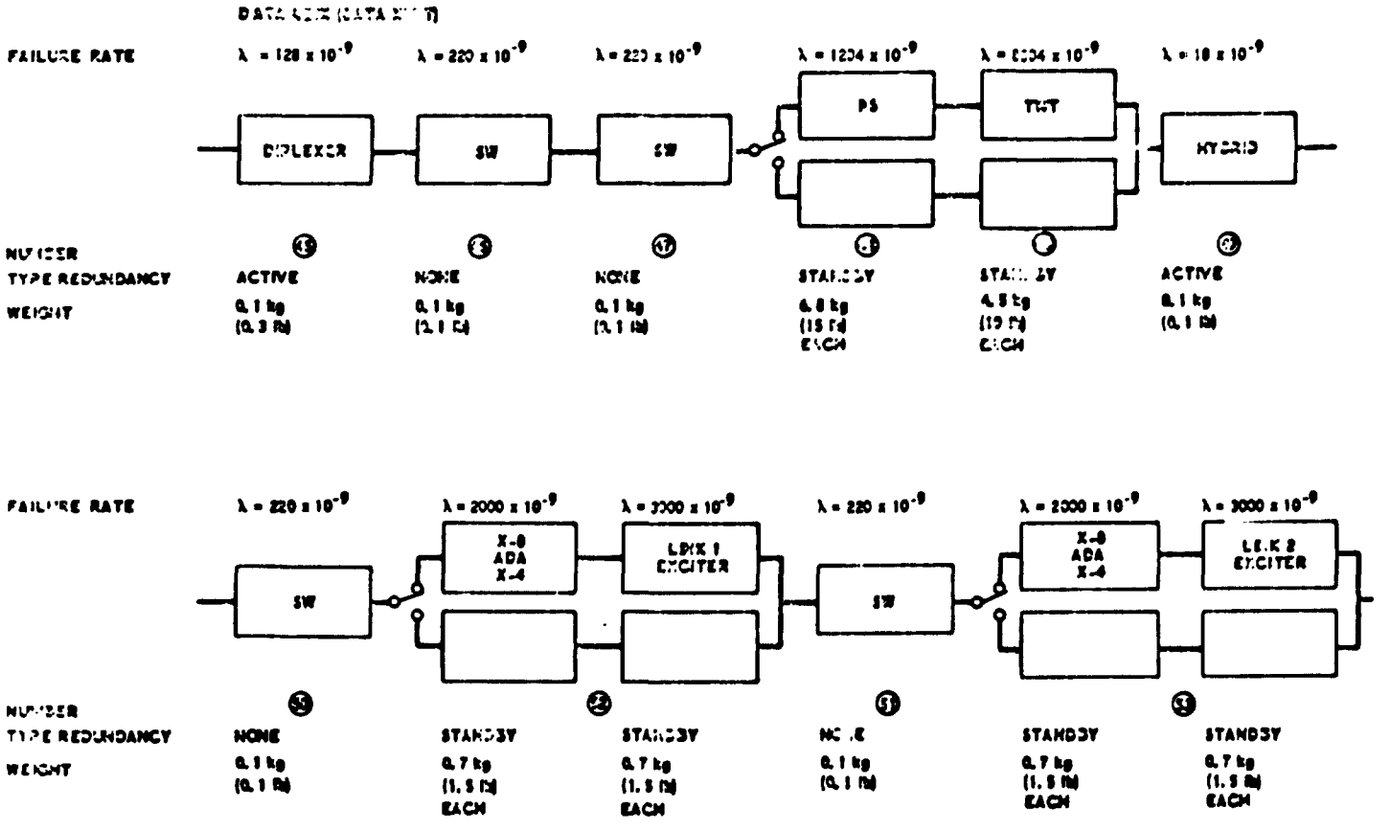


Figure 3-30. SLDSP Communication Data Transmission Reliability Model

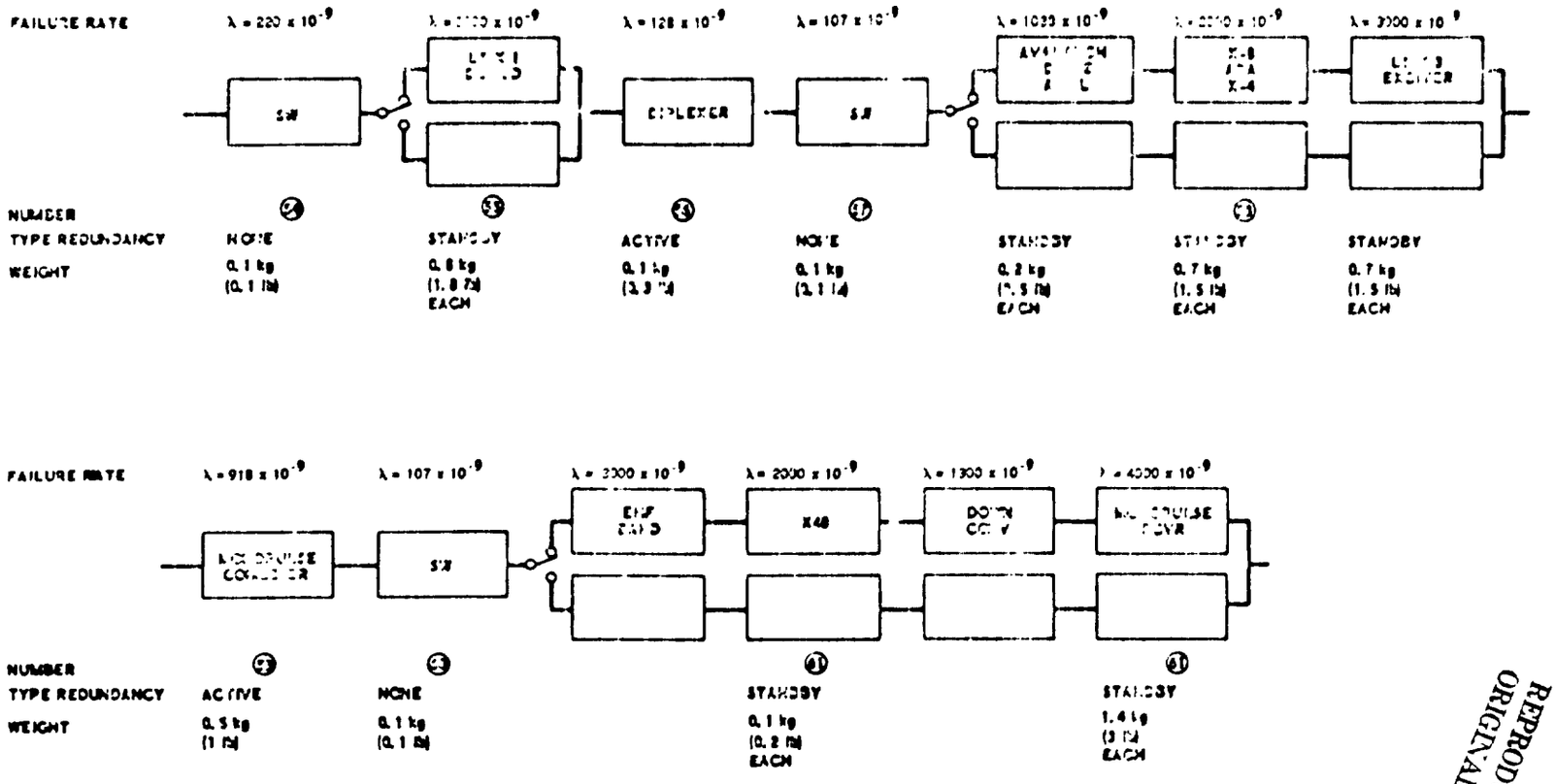


Figure 3-31. SLDSP Communication Command Transmitter and Receiver Reliability Model

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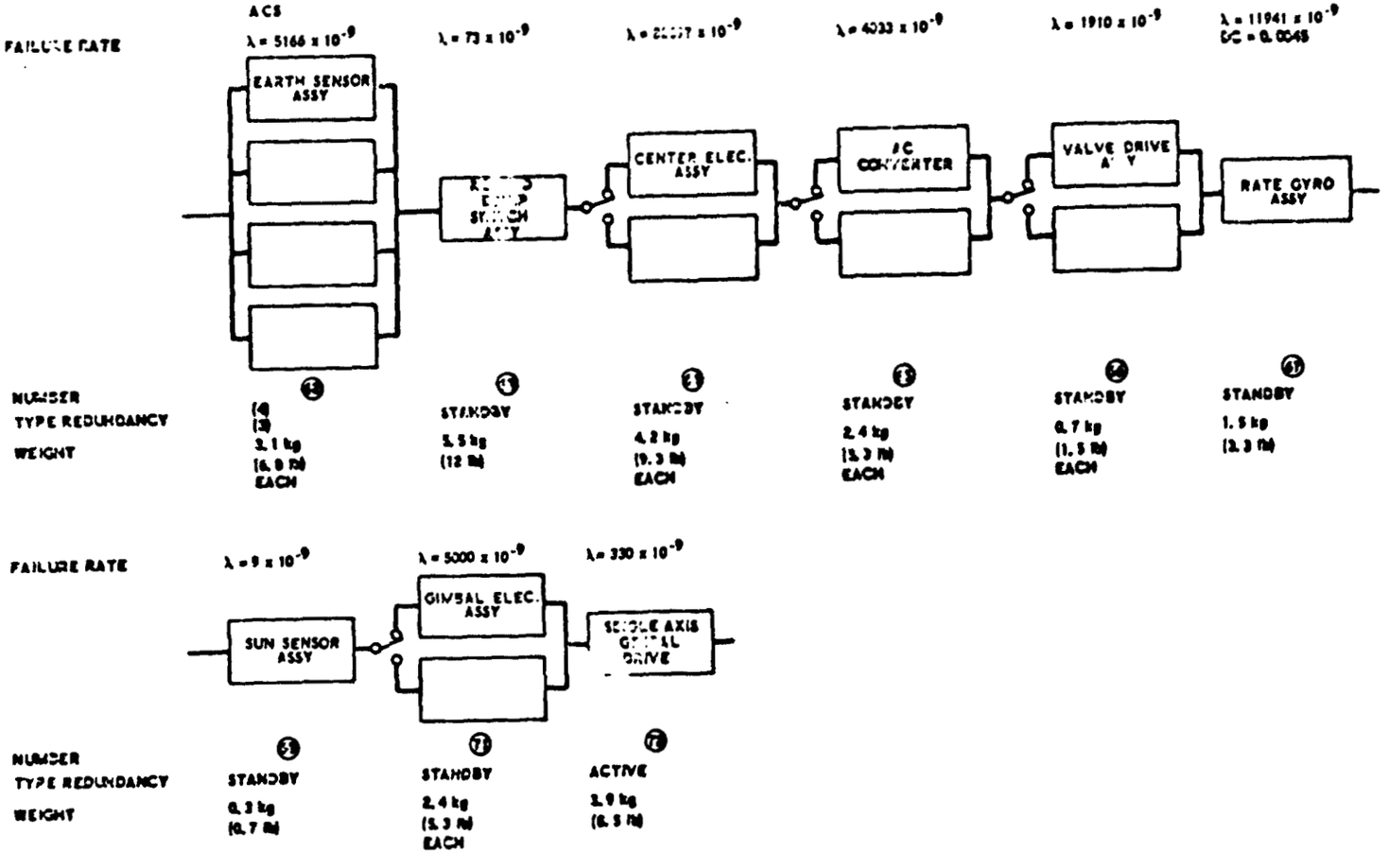


Figure 3-32. SLDSP Attitude Control Reliability Model

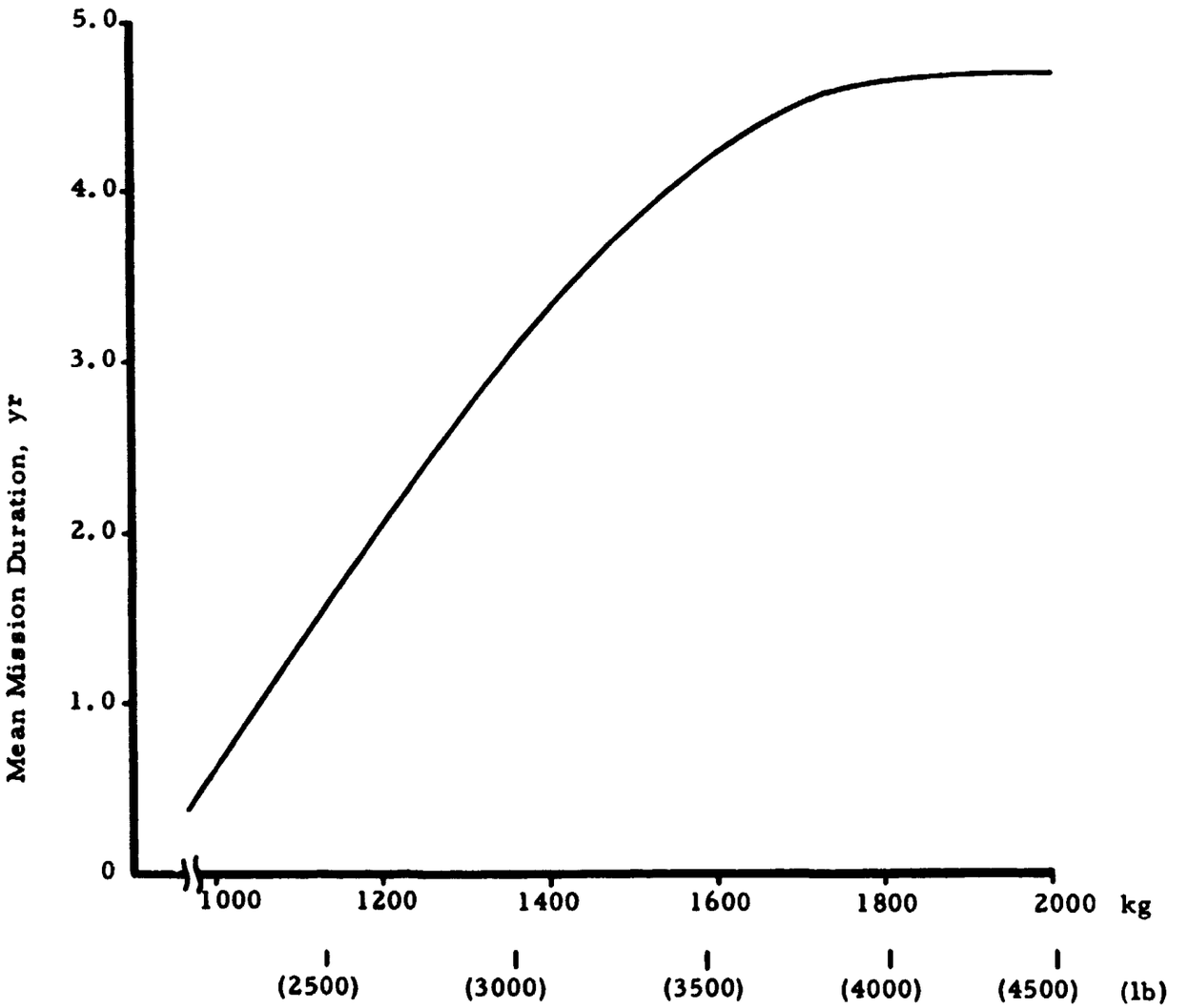


Figure 3-33. SLDSP Weight Growth with MMD

Table 3-11. SLDSP Parts Quantity List

No.	Subsystem	Module	Type Redundancy	MMD (yr) 1	2	4.5
1	Structure	Structure	None	1	1	1
2	Propulsion	Gas Generator Assy	S	1	1	2
3		Feed Assy	S	1	1	1
4		Installation Hdwr	S	1	1	2
5		Hi Level + Vernier Thrusters	S	1	2	2
6		Low Level Thrusters	S	1	1	2
7	Elec. Power	Solar Array	S	1	1	1
8		Solar Array Drive Motor	A	1	1	2
9		Shunt Element Assy	B(4)	4	4	5
10		Power Conditioning Unit	S	1	1	2
11		Reconditioner/Battery	B(2)	2	2	3
12	Elec. Distr.	Elec. Integration Assy	S	1	1	2
13		Auxiliary Integr. Assy	S	1	1	2
14		Platform Converter	S	1	1	1
15		Cabling	A	1	1	1
16	Comm (Link 1)	Hi Gain Antenna	S	1	1	1
17		Combiner	S	1	1	2
18		Convtr/Xmtr West	S	2	2	3
19		Convtr/Xmtr East	S	2	2	3
20		KGX 28	S	2	2	4
21	(Link 2)	Hi Gain Antenna	S	1	1	2
22		Combiner	S	1	1	2
23		Switch	None	1	1	1
24		Switch	None	1	1	1
25		Conv/Xmtr/DBA West	S	2	2	4
26		Conv/Xmtr/DBA East	S	2	2	4
27		KGX 28A	S	2	2	4
28		Digital Telem Unit	S	2	2	5

NOTE: A = Active
B = Binomial () # Req'd
N = Normal
S = Standby

Table 3-11. SLDSP Parts Quantity List (Continued)

No.	Subsystem	Module	Type Redundancy	MMD (yr)		
				1	2	4.5
29	Comm (Link 3)	Omni Antenna	A	1	1	2
30		Diplexer	A	1	1	2
31		Hybrid	A	1	1	2
32		Command Format	S	2	2	4
33		Combiner	A	1	1	2
34		Switch	None	1	1	1
35		Down Convtr/Uplink Portion	S	1	2	4
36		Cmd Tone Generator	S	2	2	3
37		Ranging Filter Inhibit	A	1	1	2
38		Conv & SG Cond Unit	S	2	2	3
39	Data Link (Common)	EHF & AQU Antenna	A	1	1	1
40		Dual Subdecoder	S	1	2	3
41		Dual Multiplier	S	2	2	5
42		Monopulse Comparator	A	2	2	3
43		Switch	None	1	1	1
44		Ant Structure Deploy	A	1	1	2
45	(Data Xmtr)	Diplexer	A	1	2	2
46		Switch	None	1	1	1
47		Switch	None	1	1	1
48		Power Supply/TWT	S	1	2	3
49		Hybrid	A	1	1	2
50		Switch	None	1	1	1
51		Switch	None	1	1	1
52		X8-ADA-X4 Exciter (Link 1)	S	2	2	3
53		X8-ADA-X4 Exciter (Link 2)	S	2	2	3
54		Switch	None	1	1	1
55		Link 1 Demodulator	S	2	2	3
56	(Cmd Xmtr & Recvr)	Diplexer	A	1	2	2
57		Switch	None	1	1	1
58		Avail Diode Amp/Link 3 Exiter	S	2	2	4
59		Monopulse Combiner	A	1	2	3
60		Switch	None	1	1	1
61		Down Converter	S	2	2	4

Table 3-11. SLDSP Parts Quantity List (Continued)

No.	Subsystem	Module	Type Redundancy	MMD (yr) 1	2	4.5
62	Attitude Cont	Earth Sensor Assy	B(3)	4	5	7
63		Red. Equip Switch Assy	S	1	1	1
64		Contr Elec Assy	S	2	3	5
65		A C Converter	S	1	2	3
66		Valve Driver Assy	S	2	2	3
67		Rate Gyro Assy DC-.0045	S	1	2	2
68		Sun Sensor Assy	S	1	1	1
71		Gimbal Elec Assy	S	1	2	3
72		Single Axis Gimbal Drive	A	1	1	2

analysis to examine module size and module simulation to compute the expected number of launches.

2. CONCEPTUAL DESIGN

The number of modules was varied from the eight space-replaceable modules in the reference SLDSP design reported in Reference 10. The variations in this study are:

a. Two and 13 space-replaceable module configurations

b. Parts count as determined by SYSOPT optimum parts allocation program

The basic structural concept that was defined in Reference 10 was used for the 2- and 13-module configurations. The modules in each of the on-orbit maintainable configurations were volume checked to determine if they could accommodate the equipment necessary for each of the selected MMDs (1, 2, and 4.5 years). In addition to o orbit maintainable payload concepts, a brief examination was performed on the modifications that could be made to convert to a ground-refurbishable and an expendable payload version.

a. Reference SLDSP

Figure 3-34 shows the reference SLDSP (eight modules) reported in Reference 10 and included in this report for completeness. For this study, it was desired to determine if the equipment and components required to achieve MMDs of 1, 2, and 4.5 years could be packaged in the reference concept. The parts listed in Table 3-11 were allocated to their proper space-replaceable modules, and the resulting module weights were determined. Based on an average packing density of 320 kg/m^3 (20 lb/ft^3), the required volume per module to contain the subsystems as described by the SYSOPT analysis for the MMDs of 1, 2, and 4.5 years was determined. This required volume was then compared to the available volume in the reference design. The results of this comparison are shown in the center column of Table 3-12. From the table, it can be seen that the equipment can fit within the available 0.38 m^3 (13.5 ft^3) volume per module in all cases except for the Link 2/3

This figure intentionally omitted for security reasons. See Foreword.

Figure 3-34. Space-Serviceable Spacecraft (Eight-Module Configuration)

Table 3-12. Required Volume Per Module For Subsystem Packaging

ITEM	MMD (yr)	NUMBER OF MODULES					
		2		8		13	
		$10^{-2}m^3$	ft^3	$10^{-2}m^3$	ft^3	$10^{-2}m^3$	ft^3
ACS Propulsion	1.0	1.4	0.5	1.4	0.5	0.9	0.3
	2.0	1.4	0.5	1.4	0.5	0.9	0.3
	4.5	3.1	1.1	3.1	1.1	1.7	0.6
Elect. Pwr/ Distribution (No Solar Arrays)	1.0	11.6	4.1	11.6	4.1	11.6	4.1
	2.0	12.2	4.3	12.2	4.3	12.2	4.3
	4.5	23.8	8.4	23.8	8.4	23.8	8.4
Communications Link 1 (1)	1.0	4.0	1.4	4.0	1.4	2.0	0.7
	2.0	4.0	1.4	4.0	1.4	2.0	0.7
	4.5	7.6	2.7	7.6	2.7	4.0	1.4
Link 2/(2) Link 3	1.0	15.9	5.6	15.9	5.6	15.9	5.6
	2.0	20.4	7.2	20.4	7.2	20.4	7.2
	4.5	46.1	16.3	46.1	16.3	46.1	16.3
Data Link(3)	1.0	10.5	3.7	10.5	3.7	10.5	3.7
	2.0	16.4	5.8	16.4	5.8	16.4	5.8
	4.5	28.0	9.9	28.0	9.9	28.0	9.9
ACS Elect.	1.0	9.6	3.4	9.6	3.4	4.8	1.7
	2.0	13.6	4.8	13.6	4.8	6.8	2.4
	4.5	26.0	9.2	26.0	9.2	13.0	4.6
Sensor Electronics (4)	1.0	54.0	19.0	54.0	19.0	54.0	19.0
	2.0	73.6	26.0	73.6	26.0	73.6	26.0
	4.5	120.1	42.6	120.1	42.6	120.1	42.6
Available (5) Volume/Module		566.0	200.0	38.0	13.5	28.0	10.0

NOTES:
(1) Link 1 volumes have been added to the payload volumes for total (SRU 12) required volume.
(2) Link 2 and Link 3 are in the same module.
(3) Data link equipment is on the back of the antenna. Module volume constraints do not apply.
(4) Optics weight is not included in modules.
(5) Available volumes based on a 4.4m (14.5 ft) dia x 0.76m (30 in) deep baseline spacecraft.

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communications module in the 4.5-year MMD case and the sensor electronics in nearly all cases. By increasing the packing density of the sensor electronics to 641 kg/m^3 (40 lb/ft^3) from the previously assumed 320 kg/m^3 (20 lb/ft^3), the electronics can be packaged in the sensor module in the 1- and 2-year MMD configuration. Likewise, the Link 2/3 equipment for the 4.5-year MMD can also be packaged by increasing its packing density to 641 kg/m^3 (40 lb/ft^3) from 320 kg/m^3 (20 lb/ft^3). The 641 kg/m^3 (40 lb/ft^3) packing density is considered to be a maximum and is based on densities achieved with a maneuvering reentry vehicle. The impact on the design, if this higher density were to be used, would be to reduce accessibility for installation, checkout, and repair of equipment and possibly increase equipment heating problems.

However, even with the 641 kg/m^3 (40 lb/ft^3) packing density, the sensor electronics required for the 4.5-year MMD exceed the available packing volume. There are several design solutions for this problem. One is to lengthen the spacecraft from 0.76 m (30 in.) to 2.4 m (95 in.) in order to increase the available volume per module to 1.21 m^3 (42.6 ft^3). This would allow packing of the sensor electronics in a single module at a packing density of 320 kg/m^3 (20 lb/ft^3). Lengthening the spacecraft only 1.2 m (47 in.) would allow packaging at a density of 641 kg/m^3 (40 lb/ft^3). Another solution is to divide the Link 2/3 and the sensor electronics into two modules each. This concept would have to be studied to determine the impact on the data bus design and the module/spacecraft electronics interface.

b. Maximum-Module SLDSP

The subsystems of the reference SLDSP were studied to determine the maximum number of modules into which they could be divided. With the aid of various technical specialists, the subsystems were divided into 13 modules and arranged as shown in Figure 3-35.

It should be noted that there are spaces for 16 modules, beginning with No. 1 in the 1:00-o'clock position and preceeding clockwise to No. 16. Spaces 1, 2, and 15 are left empty to provide a clear view for the second-surface mirror radiator on the sensor (module No. 16). Spaces No. 1 and 15, however, contain second-surface mirrors to provide a cold viewing surface for the radiator.

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security reasons. See Foreword.

Figure 3-35. SLDSP Spacecraft (13-Module Configuration)

Modules 3, 5, 11, and 13 contain the attitude control propulsion. The propulsion subsystem was divided into four self-contained modules in order to gain a longer control moment arm about the roll axis. They were placed in slots 3, 5, 11, and 13 rather than the 45-deg slot positions 2, 6, 10, and 14 in order to avoid interference with the second-surface mirrors of the sensor and also to allow for placement of the cross-link antenna in slot No. 6.

Module No. 4 contains a completely redundant Link 1 data transmission system including a replaceable antenna. The Link 1 system is completely redundant, rather than partially redundant, to avoid a large impact and major redesign on the data bus and also to avoid a multi-pin connection at the module/spacecraft interface.

Module No. 6 contains the cross-link communications system. To reduce power losses and to eliminate waveguide interfaces, the electronics were placed on the back of the antenna. This system could not be split-up into more than one module due to the large power losses which would be incurred with long lengths of waveguide.

Modules No. 7 and 10 contain completely redundant attitude control electronics and sensors. The RADEC systems were split between modules No. 8 and 9 because their functions are distinct and somewhat different.

Module No. 12 contains the Link 2 and 3 communications system. This system could not be split into more than one module because of the impact on the design of the data bus. Also, the basic single-pin module/spacecraft interface described in Reference 10 would have to be increased to a multi-pin connector should the Link 2/3 system require splitting.

The electrical power system, consisting of a single array and batteries, was left in one module. Attempts to split the array into two resulted in mutual array shadowing and interference with the line of sight for the radiators.

Module No. 16 contains the sensor, the second Link 1 system, and the Data Processing Electronics System. It is uncharged from that described in Reference 10.

Table 3-11 lists the parts count provided by SYSOPT. The Link 1 systems are located in modules 4 and 16 and were designed to be completely redundant systems; i.e., the system in module 4 is to be identical to that in module 16. However, the equipment list was changed to show a single antenna and some electronics in module 16 and the remainder of the Link 1 electronics in module 4. The impact of this equipment split on the design is not known. It is quite possible that it would require multi-pin connectors at the module/spacecraft interface, and perhaps require that the data bus be redesigned.

The equipment required for achieving MMDs of 1, 2, and 4.5 years was checked to determine if it could be packaged within the module as shown in Table 3-12. As with the 8-module configuration, problems arose with the 13-module configuration. In this case, the spacecraft would have to be lengthened to 3.25 m (128 in.) to accommodate the sensor electronics in a single space-replaceable module package at 320 kg/m^3 (20 lb/ft^3) or lengthened to 1.62 m (64 in.) to package it at 641 kg/m^3 (40 lb/ft^3). The design possibilities for this problem would be the same as those suggested for the eight-module configuration.

c. Minimum-Module SLDSP

Figure 3-36 is an outboard profile of a SLDSP spacecraft configured with two modules. Although the shape of the 2-module payload is somewhat different than that of the 8- and 13-module versions, the baseplate concept and the index/translation motion for removal/replacement of the module is retained. The configuration was developed to determine the distribution of equipment for each module for use in the reliability model. There are no problems involved with packaging the equipment required for MMDs of 1, 2, and 4.5 years into this spacecraft as shown in Table 3-12.

d. Ground-Refurbishable and Expendable SLDSPs

A ground-refurbishable version of the DSP, based on the concept developed for the on-orbit maintainable SLDSP, would be identical to the version shown in Figure 3-34 except that the equipment required for on-orbit servicing would be removed. This would include the baseplates, tracks, and

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security reasons. See Foreword.

Figure 3-36. SLDSP Spacecraft Two-Module Configuration

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perhaps the data bus. The baseplates would be replaced with the necessary bracketry for equipment mounting tied directly to the semi-monocoque structure. Also, the replaceable drives would be replaced with conventional shaft connections.

The redundancy requirements to achieve MMDs of 1, 2, and 4.5 years would not result in the same packaging problems as for the space-serviceable versions, because the envelope constraints of the individual module no longer apply since the modules are no longer removable. Excess equipment may be mounted on the inside of the main structural ring to increase the envelope available for equipment packaging.

The expendable SLDSP is identical to the ground-refurbishable SLDSP except that, in addition to the removal of the servicing equipment, the docking cone is also removed. Here again, an increase in usable volume for the equipment packaging is available on the inside of the main structural ring.

Due to brevity of the study, the ground-refurbishable and expendable versions of the SLDSP were assumed to be simplified versions of the on-orbit maintainable version for comparative purposes. It should be noted, however, that if either concept proves attractive, specific configurations should be developed which would be optimized for either the ground-refurbishable or expendable SLDSP payload rather than using configurations based on an on-orbit maintainable SLDSP.

3. MASS PROPERTIES

The reference SLDSP weight estimate was developed from the Spacecraft Synthesis Program (SSPRO) using the mission equipment defined in Reference 10. The SLDSP reference weight is for an expendable payload that is three-axis stabilized with a one-year MMD. The current expendable satellite is a yaw-spin configuration. The major differences between the current and the reference configuration are the changes to the mission equipment, stabilization concept, and structural design to use the full 4.4-m (14.5-ft) diameter, and the elimination of the movable ballast.

With these modifications to the configuration for on-orbit servicing, the reference weight was determined using the SSPRO. A printout of one

SLDSP case is duplicated in Table 3-13 which includes the weight estimates for the reference weight and three Shuttle payload types. It should be recognized that for the case examined in Table 3-13, the reference payload weight is for a one-year MMD and the three Shuttle payloads are for two-year MMDs.

To account for the MMD effect on weight, the parts count determined by the SYSOPT program was used. The resulting weight relationship is provided in Figure 3-37 for the subsystems including the mission equipment. The influence of modularity was determined from several payload modularity studies. These data were shown in Figure 2-4. With the MMD and modularity relationships, the SSPRO computed the weights as MMD was varied from 1 to 4.5 years and modularity was varied from 2 to 13 space-replaceable modules.

4. EXPECTED NUMBER OF LAUNCHES

The expected number of launches for the SLDSP was determined from the SAMS program for configurations with 2, 8, and 13 modules and MMDs of 1, 2, and 4.5 years. The Weibull functions for each of these modules were computed from the reliability model. These functions are listed in Table 3-14 with an estimate of the predicted MMD. These were inputted into the SAMS to compute the expected number of launches for the three types of replacement strategy (random, conditional, and perfect failure prediction) which were considered in evaluating the method of selecting the modules to be replaced beyond the failed module. The expected numbers that are shown in Figures 3-38, 3-39, and 3-40 are applicable to on-orbit servicing of space-replaceable modules. The non-replaceable module consists of the spaceframe, cables, and sensor optics, and any failures in these areas will require ground refurbishment. The expected number of launches per year for the non-replaceable items is 0.014 for all of the on-orbit maintainable configurations.

5. COST ANALYSIS

The unit cost of the SLDSP was computed for expendable, ground-refurbishable, and on-orbit maintainable payloads with MMDs of 1, 2, and 4.5 years. These costs are shown in Figure 3-41. Also shown in the figure is the total program cost of a one-SLDSP on-orbit system for a 10-year

Table 3-13. SSPRO Weight Prediction for the SLDSP Payload

PROGRAM	SSPRO				
CASE NUMBER 1	SLDSP	SATWTS			07/12/73
MISSION LIFE (YR.)	10.		MEAN MISS. DUR.		2.000
DESIGN LIFE (YR.)	3.600				
SATELLITES IN SYSTEM	1.		MODULARITY (UNITS)		8.
MISS. POWER (W)	728.		BATT. REDUNDANCY		0.
POINT. ACCUR. (DEG)	.450		PAYLOAD DENSITY		3.0
TYPE ELECT. POWER	SOLAR	ORI	FLEX		
I. O. C.	1380.		VELOCITY		0.
ARRAY PACK. FACT.	.3		EXT. DN. LINKS		2.
NUMB. TAPE RECORD.	2.0		DATA PROCESS. WT.		190.
ENCRYPTION WT.	0.				

ITEM	MMD FACTOR	GR MOD FACTOR	OM MOD FACTOR	REFERENCE WEIGHT	EXPEND SHUTTLE	GROUND REFURB	ON-ORB MAINT.
STRUCTURE	1.000	1.000	1.000	191.	149.	158.	310.
ENVIRON. CONT.	1.000	1.000	1.100	48.	73.	67.	82.
GUID. NAV. + STAB.	1.185	1.000	1.000	114.	135.	145.	145.
DRY PROPULSION	1.000	1.000	1.000	0.	0.	0.	0.
REACT. CONT.	1.104	1.000	1.000	26.	53.	53.	61.
C. D. P. I.	1.235	1.000	1.000	244.	301.	301.	301.
ELECTRICAL	1.000	1.000	1.000	519.	529.	529.	567.
SOLAR ARRAY			0.000	90.			
BATTERY			0.000	120.			
DISTRIBUTION			0.000	229.			
PWR. CONDIT.			0.000	80.			
MISS. EQUIPMENT	1.187	1.000	1.000	518.	615.	615.	615.
CONTINGENCY	0.000	0.000	1.000	178.	192.	194.	243.
DRY WEIGHT				1826.	2044.	2063.	2423.
REACT. CONT. PROPEL.				84.	171.	173.	203.
MAIN PROPELLANT				0.	0.	0.	0.
WET WEIGHT				1910.	2215.	2236.	2626.
ADAPTER WEIGHT	1.000	0.000	0.000	126.	120.	212.	231.
PALLET WEIGHT	1.000	0.000	0.000	0.	0.	0.	0.
LAUNCH WEIGHT				2036.	2335.	2448.	2857.

Table 3-13. SSPRO Weight Prediction for the SLDSP Payload (Continued)

PAYLOAD GEOMETRY							
DIAMETER (FT.)				8.5	14.7	14.7	14.7
LENGTH (FT.)				9.5	3.4	3.6	4.1
ADAPT. LENGTH (FT.)				.8	.6	.6	.6
DENSITY (LB/CU FT)				3.8	3.8	3.8	3.8
TOTAL ELECT. POWER (W)				928.0	928.0	928.0	928.0
BOOSTER DIAM. (FT)				10.0	13.5	13.5	13.5
MEAN MISS. DUR.				1.000	2.000	2.000	2.000
DES. LIFE				1.000	3.60	3.600	3.600
COST WEIGHT SUMMARY							
				REFERENCE WEIGHT	EXPEND SHUTTLE	GROUND REFURB	ON-ORB MAINT.
STR + TPS + ADP	WS			390.6	339.2	437.4	621.8
GN + ACS	WA			162.2	188.0	198.4	206.1
ORV PROPULSION	WP			0.0	0.0	0.0	0.0
C. D. P. I.	WO			281.5	301.0	301.0	301.0
ELECTRICAL	WE			600.0	529.2	529.2	657.1
MISSION EQUIP.	WM			918.0	614.8	614.8	614.8
ATT. CONT. PROP.	WAP			84.1	171.5	173.1	203.4
MAIN PROPELL.	WPP			0.0	0.0	0.0	0.0
TYPE STRUCTURE	S1 =	2.	EXO				
ELECT. POWER (WATTS)	E1 =			928.	928.	928.	928.
ORBIT ALTITUDE	O1 =	24000.					
TYPE STABILITY	A1 =	3.	3- AXIS				
TOTAL IMPULSE	P1 =			0.	0.	0.	0.
TYPE PROPELLANT	P2 =	4.	LIQUID				
LAUNCH VEHICLE		1.	SHUTTLETUG				
TYPE MISS. EQUIP.			COMPLEX				

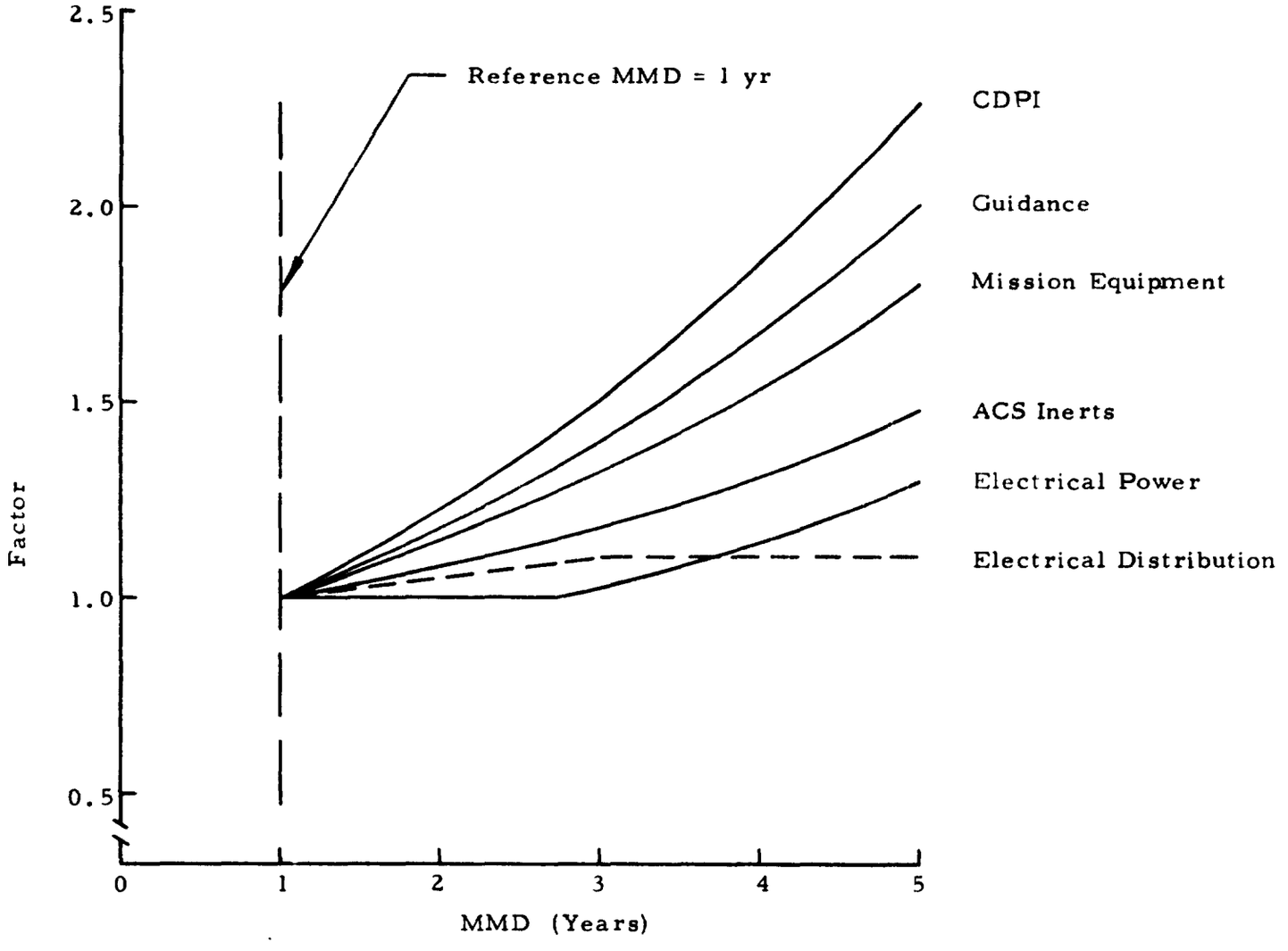


Figure 3-37. SLDSP Weight Factors Versus MMD

Table 3-14. Weibull Functions for SLDSP Modules

Configuration	Module Number	α (Mo.)			β			MMD (Mo.)		
		1 Yr	2 Yr	4.5 Yr	1 Yr	2 Yr	4.5 Yr	1 Yr	2 Yr	4.5 Yr
2 Modules	1	29.76	82.42	80.86	1.264	1.252	2.781	36	NA	NA
	2	35.59	43.83	113.3	1.345	1.289	1.778	36	NA	NA
	System	18.71	31.73	63.5	1.291	1.264	2.215	12	24	54
8 Modules	3, 9	311.4	1860.0	4587.0	1.0	1.01	1.121	32	42	60
	8	130.8	130.7	336.0	1.0	1.0	1.886	79	79	114
	12	28.3	60.77	80.4	1.46	1.80	2.86	26	54	71
	7	76.2	76.21	251.2	1.642	1.695	2.241	65	65	113
	4	101.8	122.9	134.5	1.60	1.611	1.80	78	86	92
	5	77.6	108.0	166.0	1.68	2.108	1.985	66	86	102
	6	1.0 E5	1.0E5	1.0E5	1.0	1.0	1.0	120	120	120
	System	18.25	30.72	65.0	1.398	1.518	1.90	12	25	54
13 Modules	3, 5, 11, 13	1627.0	3717.0	7354.0	1.0	1.011	1.175	36	42	60
	14	130.8	130.7	336.0	1.0	1.0	1.886	79	79	114
	16	28.3	66.5	80.4	1.46	1.602	2.86	26	58	71
	6	101.8	122.9	134.5	1.60	1.611	1.80	78	86	92
	4	280.9	257.7	789.6	1.831	1.921	2.225	112	111	120
	12	76.2	76.2	251.2	1.642	1.695	2.241	65	65	113
	10	115.4	152.0	222.1	1.683	2.40	3.708	85	103	118
	7	118.5	145.5	189.6	1.760	1.975	1.841	87	97	104
	8	1.5E5	1.5E5	1.5E5	1.0	1.0	1.0	120	120	120
	9	1.0E9	1.0E9	1.0E9	200.0	200.0	200.0	--	--	--
	System	18.85	30.36	61.0	1.407	1.443	2.1	13	24	52

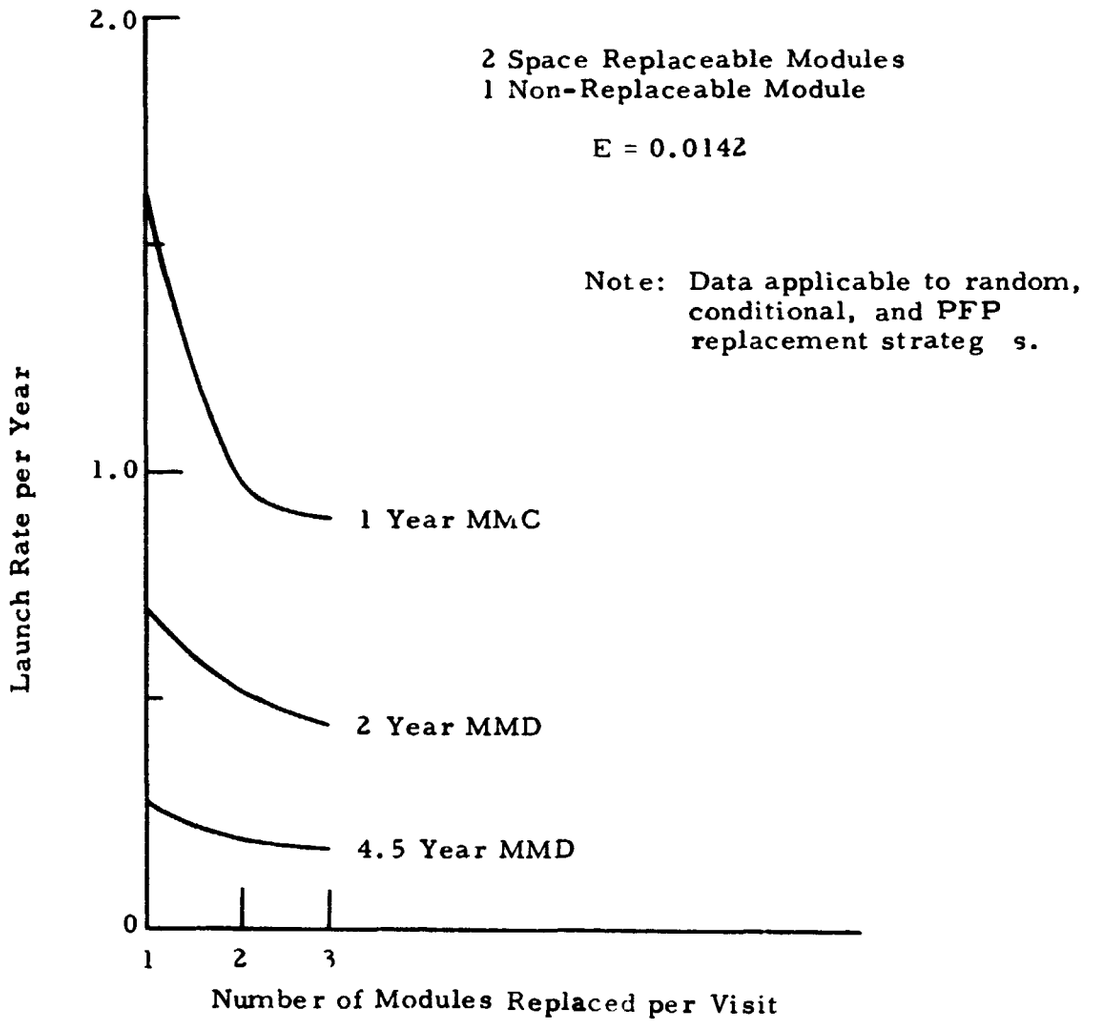


Figure 3-38. Expected Launch Rate for a Two-Module SLDSP

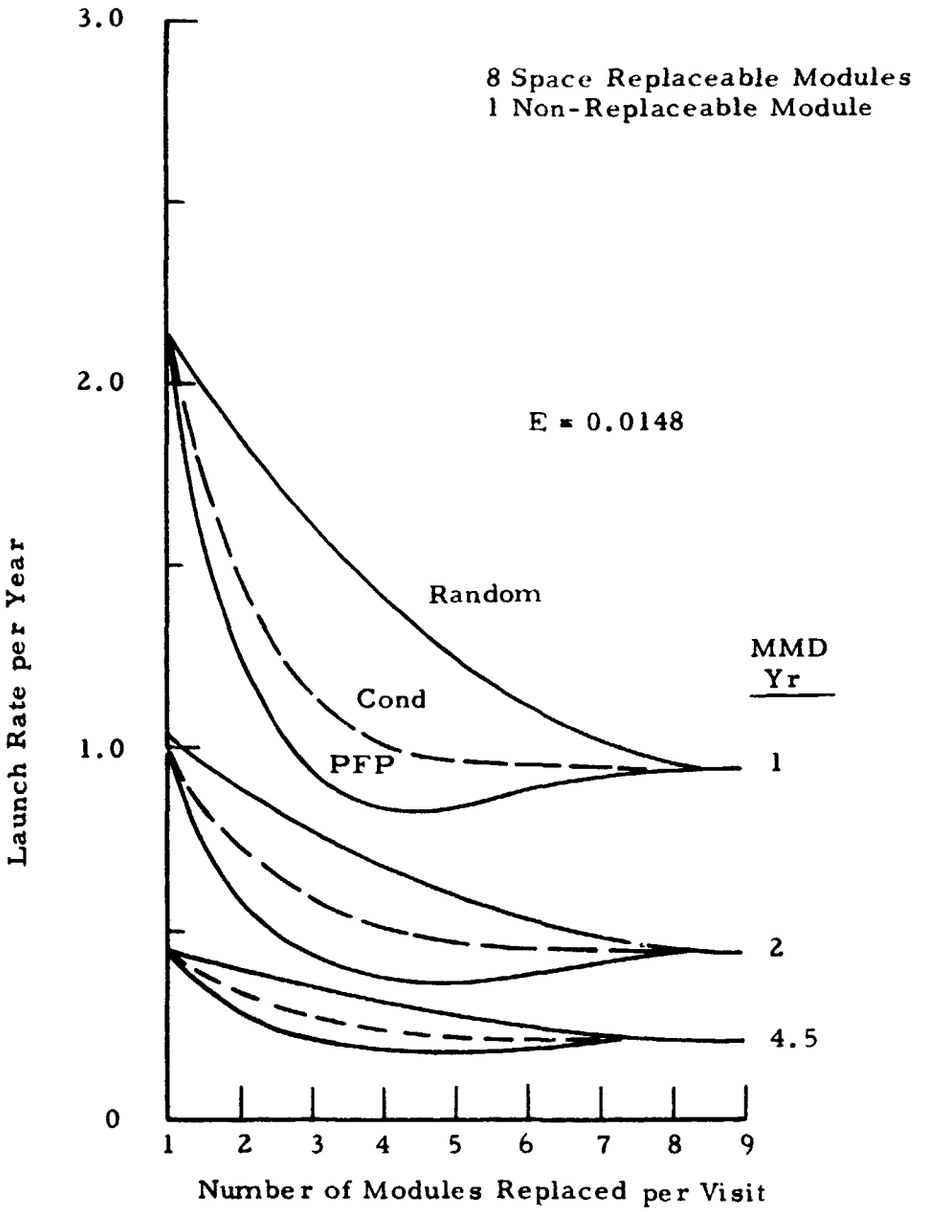


Figure 3-39. Expected Launch Rate for an Eight-Module SLDSP

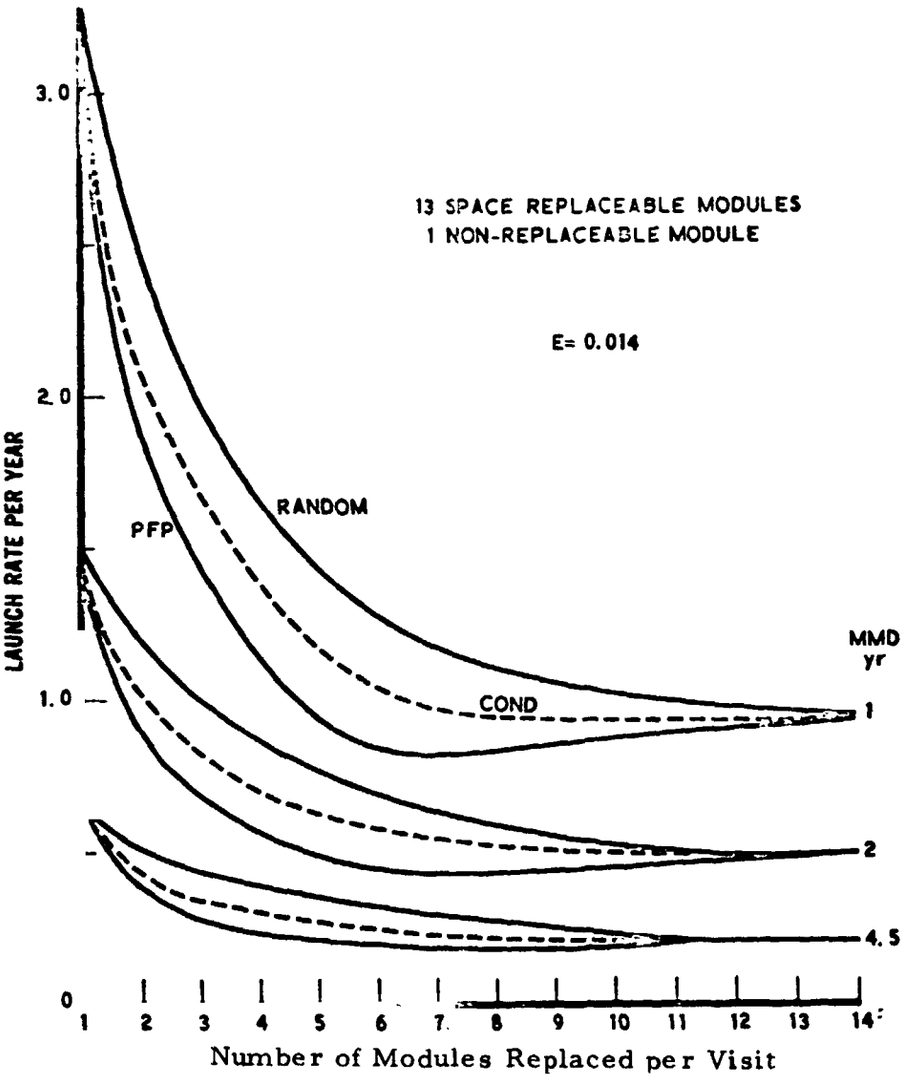


Figure 3-40. Expected Launch Rate for a 13-Module SLDSP

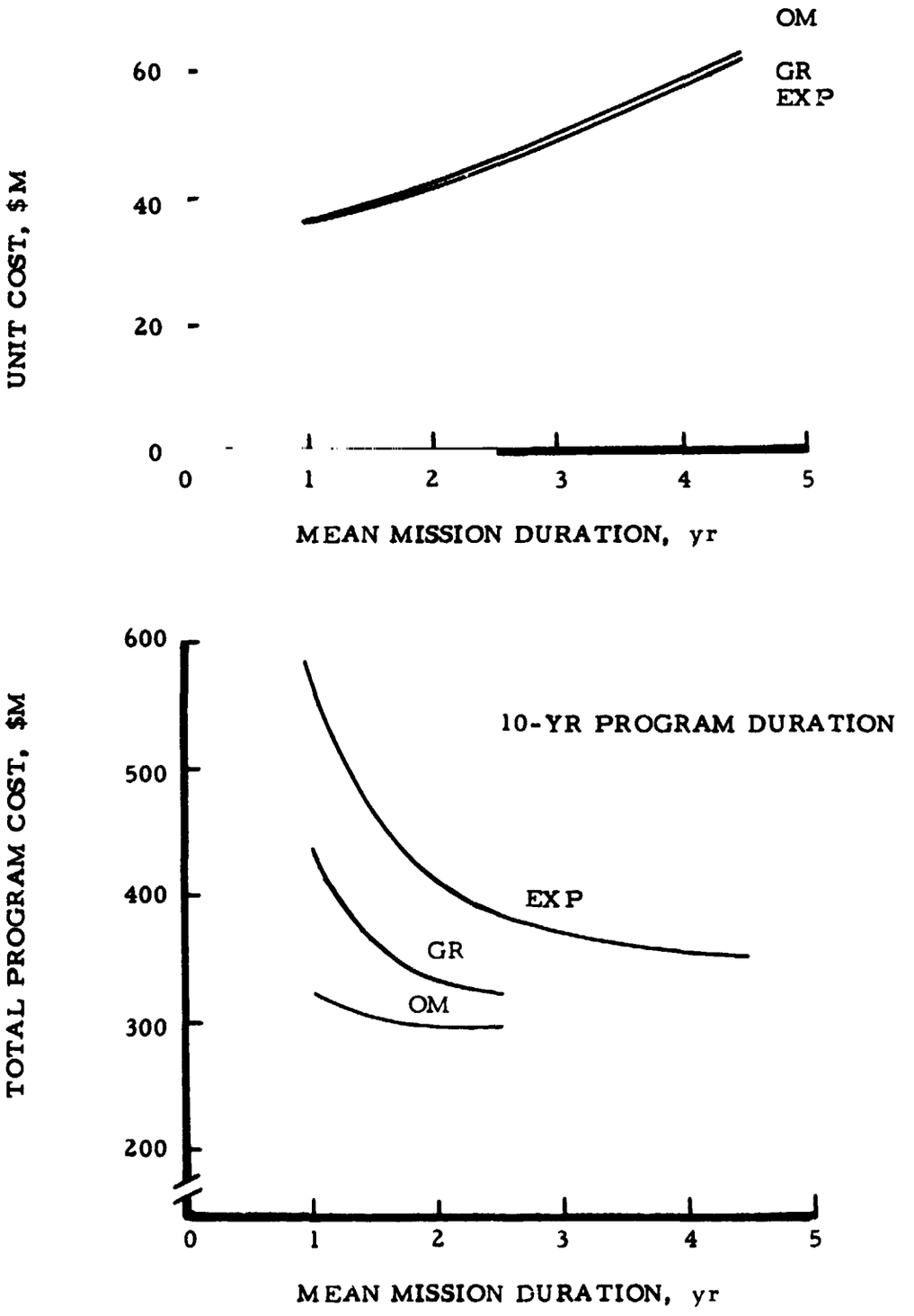


Figure 3-41. Effect of Mean Mission Duration on SLDSP Unit and Program Cost

program duration. The total program cost includes payload unit, operations, and RDT&E cost. The on-orbit maintainable and ground-refurbishment payloads appear to reach a minimum program cost, but the total program cost of the expendable payload continues to decrease as MMD is increased. At a 4.5-year MMD, the ground-refurbishable and on-orbit maintainable payloads are not expected to require refurbishment because the spare payloads will service the expected failures. It is not economical to design payloads with MMDs beyond what can be effectively used during the program duration unless high reliability is desired. The optimum MMD for on-orbit servicing of a one-satellite system appears to occur at about two years and it should be more economical than a 4.5-year expendable payload. Sortie payload is not included in this comparison because the mission operates at high altitude.

The repair strategy for the configurations with the 2, 8, and 13 modules was performed by varying the number of modules replaced per visit. The results of this tradeoff are shown in Figures 3-42 and 3-43 which reflect the performance limits of the Tug. For the 8- and 13-module designs the 4- and 6-module replacements, respectively, exceed the Tug service-sharing capability. The Tug is performing dedicated service missions when more than four and six modules per payload are replaced. The minimum cost however, is reached prior to exceeding the Tug performance capability. The optimum appears to be replacement of about 3 and 4 modules for the 8- and 13-module configurations. This amounts to about 30 percent of the total number of space-replaceable modules that should be replaced per visit.

The cost effect of the number of modules in a payload on total program cost is shown in Figure 3-44 for designs with MMDs of 1, 2, and 4.5 years. Both the 8- and 13-module configurations reflect the 30-percent replacement concept; however, the 2-module configuration cannot show this effect because any servicing will replace 50 percent or 100 percent of the modules. The 4.5-year MMD design does not show this pattern for a two-module configuration since there is no maintenance performed on the replaced modules. This tradeoff indicates that the number of modules in the payload

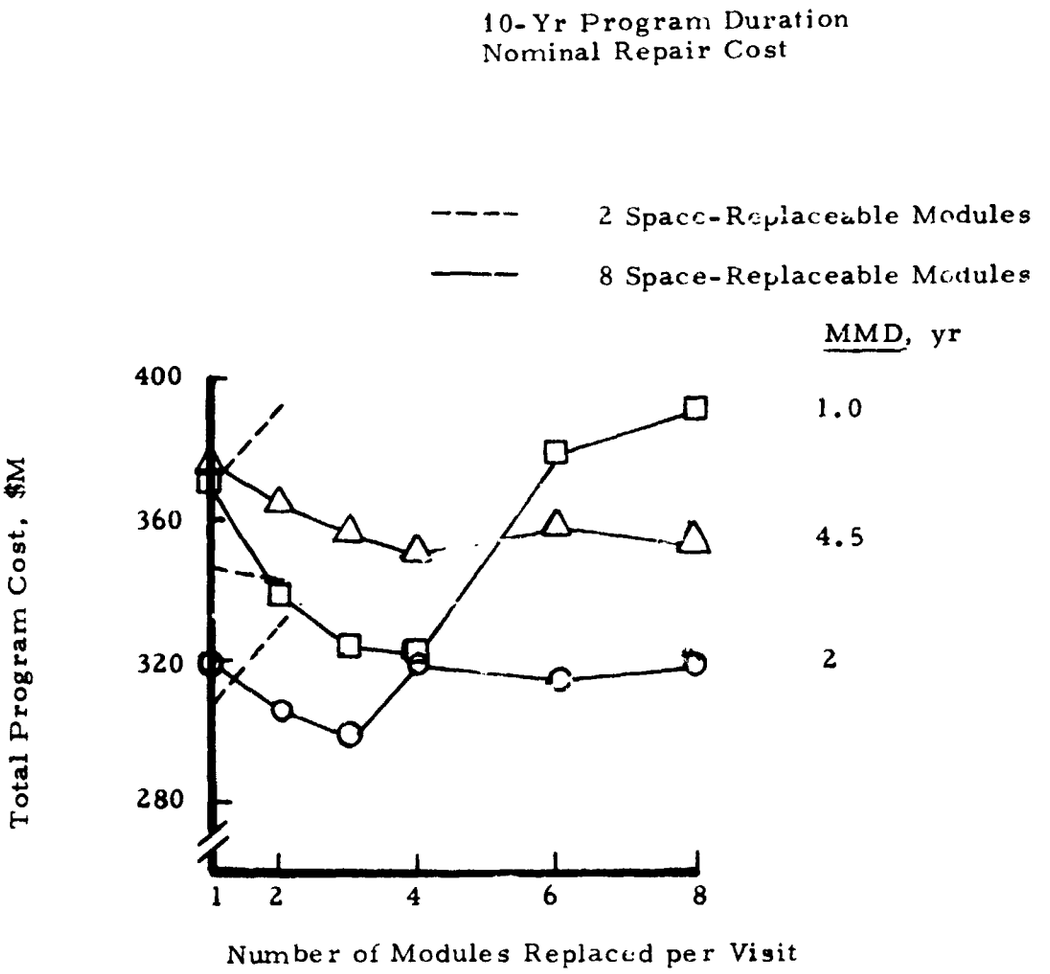


Figure 3-42. Effect of Modules Replaced on Total Program Cost for SLDSP with Two and Eight Space-Replaceable Modules

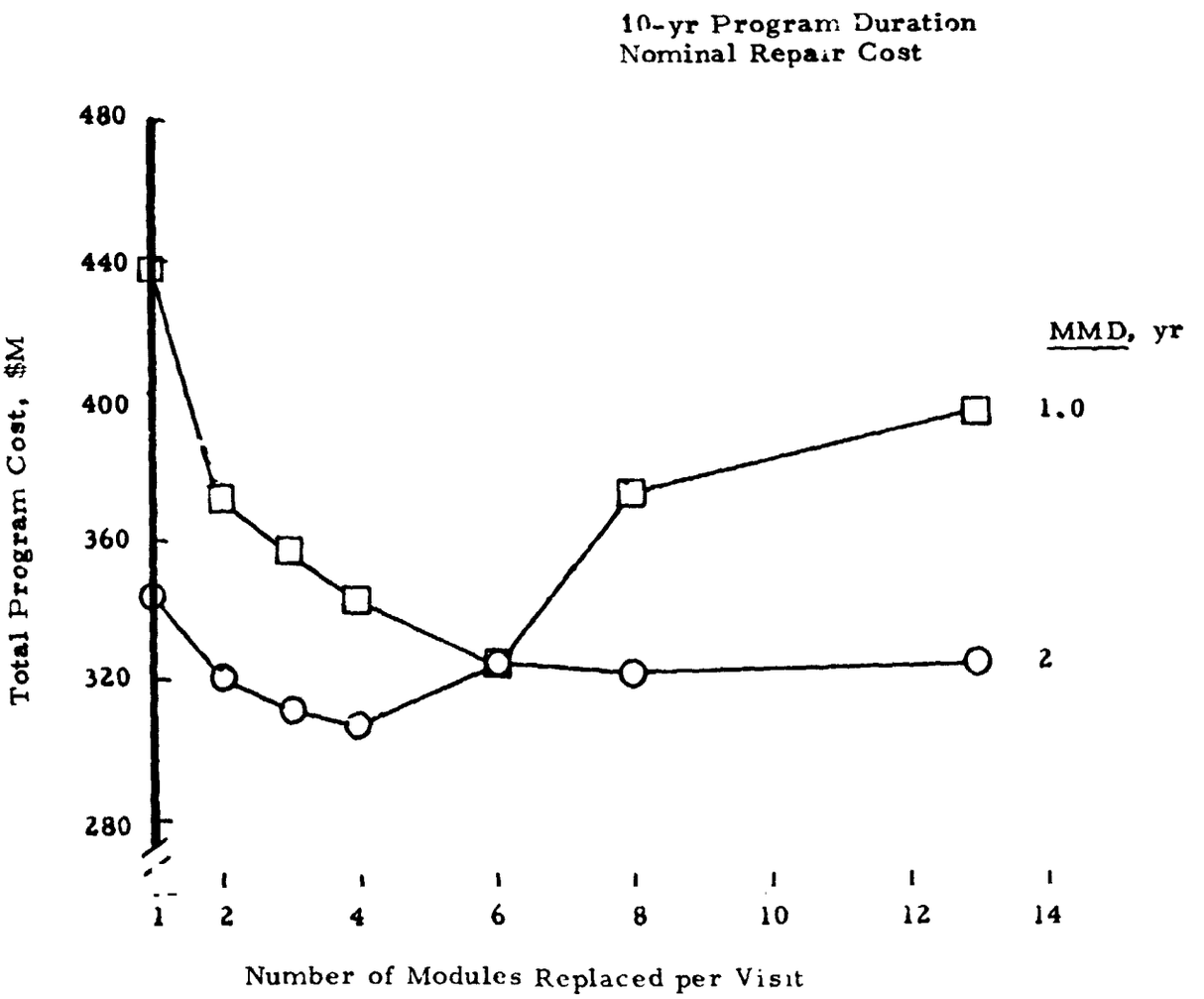


Figure 3-43. Effect of Modules Replaced on Total Program Cost for SLDSP with 13 Space-Replaceable Modules

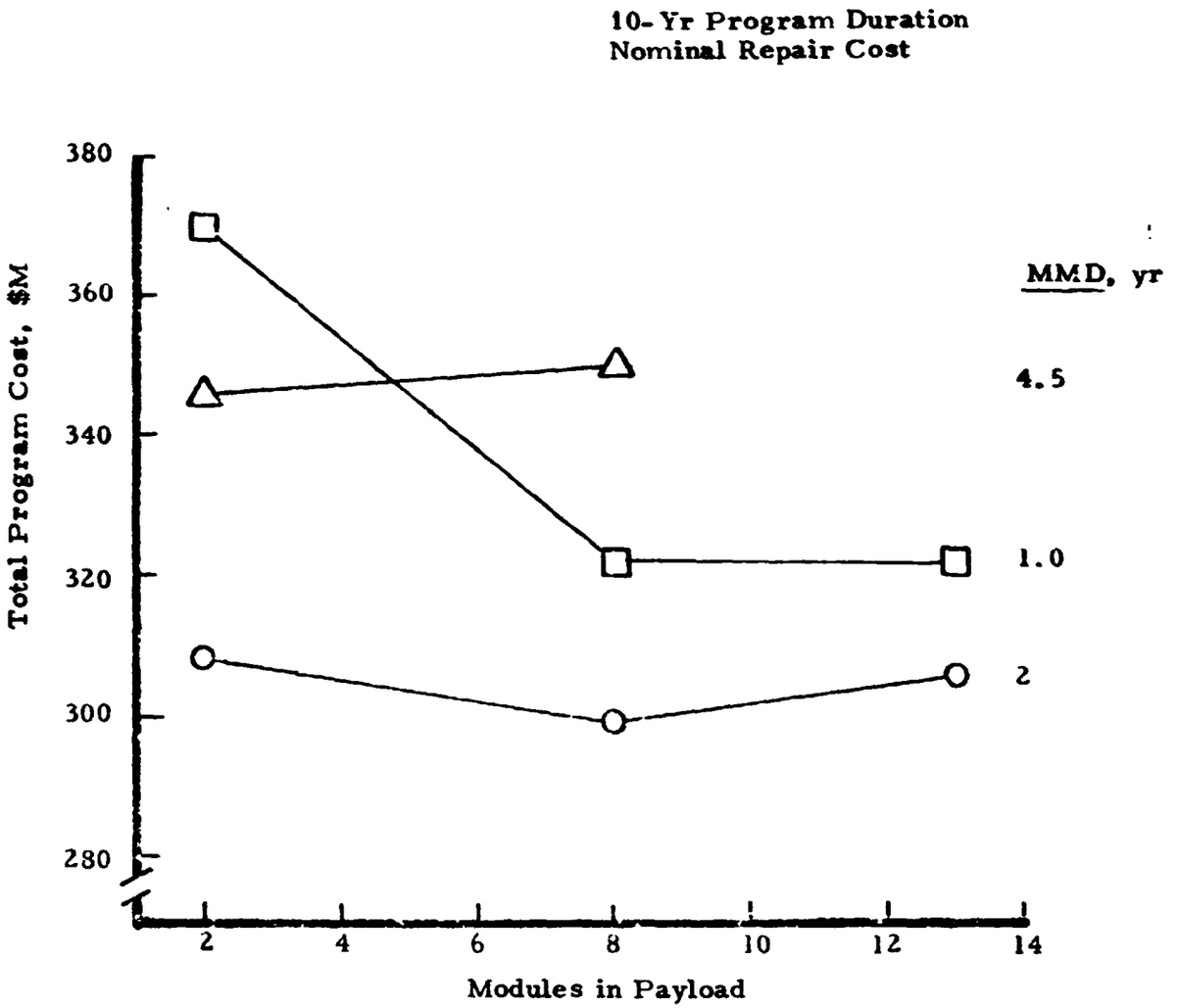


Figure 3-44. Effect of Number of Modules in SLDSP on Total Program Cost

should be enough to effectively utilize the 30-percent optimum replacement for a one-satellite system. Operational satellites which generally have several satellites in the system may exhibit a different optimum replacement percentage. The multi-payload case should be investigated in any follow-on study.

4. COMMON HARDWARE

A. INTRODUCTION

The results of two studies are presented which indicate the potential savings in costs that may result from the utilization of common attitude control subsystem hardware and of standardized solar cell modules. Commonality of spacecraft hardware should, of course, be a very powerful means of reducing payload costs. Also, for the concept of on-orbit service and repair of spacecraft subsystems by the Orbiter or Tug to be feasible, it is necessary to have equipment in the form of easily replaceable modules, and it is essential to standardize these modules to the greatest extent possible for refurbishment.

In order for a single attitude control design to be suitable for a group of vehicles having different values for the design parameters which determine stabilization requirements, a common specification bounding the requirements must be met by the subsystem. That is, it must satisfy the mission requiring the greatest accuracy, the one with the longest design life, the one with the heaviest vehicle, etc. This implies that for some, if not all, of the missions, the equipment is overdesigned and, therefore, is probably larger and heavier than subsystems would be if they were custom designed for each particular application. However, overdesign can be viewed as performance margin and, as will be discussed presently, may prove to be advantageous if held within reasonable bounds. Furthermore, in view of the Shuttle's capability for placing very heavy payloads into orbit, the importance of small size and weight can be de-emphasized.

At every step, from the original conceptual design to the end of the operational phase, commonality of subsystem hardware can provide large cost savings on a space program involving a number of different missions (Table 4-1).

Table 4-1. Cost Savings from Common Hardware

ACTIVITY	REASONS FOR LOWER COST
Preliminary System Studies	Fewer studies
Preliminary Development	Fewer development areas
Preliminary and Final Design	Fewer designs
Drawings	Fewer drawings
Tooling	Common tooling
Fabrication	Lower unit costs due to large quantities of same parts
Acceptance Test	Common test specs, test procedures, and test equipment. Simplified training of test personnel
Final Development	Fewer designs to debug
Installation	Common fixtures. Simplified training of installation personnel
Pre-flight Checkout	See Acceptance test, above
Maintenance and Repair	Common tools, instructions, procedures and spare parts. Simplified training of maintenance and repair personnel

The performance margin provided by designing the equipment to the "worst-case" specification, covering a group of space vehicles with varying requirements, can be a valuable feature. For example, assume that on a group of 12 different vehicles, 3 require a pointing accuracy of 0.002 radians while 9 others need only 0.01 radians. The attitude control subsystem would then have to be designed around the 0.002-radian requirement, which is five times as accurate as it needs to be for three quarters of the flights. However, on these nine missions, there is a five-to-one performance margin which means that the equipment accuracy could deteriorate in service by a factor of five and still meet specified pointing requirements. This should significantly improve the reliability and life of the subsystem.

Another possibility for exploiting performance margin is to have two classes of test specifications for the same equipment; one giving requirements for high-accuracy missions and another for the less-accurate missions. On the latter equipment, the performance margin could significantly reduce test costs. Subsystems presently in production appear to be designed with little performance margin, which is probably due to the current emphasis on small size and weight. Unless they are "tuned up" with each assembly at peak performance, they are likely to fail certain tests. As a result, months may be wasted nursing such equipment through test since many adjustments, replacements, and reruns must be made. In addition, much time and effort may often be devoted to negotiating waivers with the procuring agency.

The use of two sets of specifications must be traded off against other considerations, such as the necessity for having two types of spare subsystems as an example. Also, caution is advisable when equipment shows errors during test which are significantly larger than those permitted by the high-accuracy specifications. This may indicate a malfunction which will seriously jeopardize the reliability on the moderate-accuracy applications.

Commonality results in larger quantities of each item and should also increase reliability, since larger production runs usually produce more reliable equipment. Most of the "bugs" are disclosed on early lots, and appropriate action has already been taken to remove them before the majority of the units are manufactured. Many of these problems are not design deficiencies but are in the area of quality control. As the program proceeds, modified manufacturing or assembly operations and additional inspection procedures can be initiated to prevent recurrence of these problems.

It appears that the policy of common hardware should be adopted to the greatest possible extent on payload subsystems. This approach should result in very significant cost savings in engineering, manufacturing, test, maintenance, and repair and should provide hardware having improved reliability.

B. ATTITUDE CONTROL HARDWARE

The spacecraft stabilization requirements for 53 different payloads were determined from the 1971 and 1972 Shuttle Payload Data Books (Ref. 11 and 12). A total of 177 flights is planned for these payloads, some involving as many as 12 flights.

The 53 payloads were arranged in 14 groups in which the mission stabilization requirements and spacecraft configuration appear sufficiently similar to permit common attitude control subsystems to be employed without considering design life. It was assumed in this study that mean mission duration times for on-orbit or retrieval servicing intervals can be achieved by redundancy of the same stabilization assemblies.

Since all Sortie missions covered in the 1972 NASA Payload Data Book are stated to require no stabilization subsystem (because attitude control will be provided by the Orbiter), the question of stabilization hardware commonality on that type of mission was not addressed.

1. ANALYSIS

The 53 payloads considered in this study were taken from the NASA and Civil Payload Data Summary. This summary lists 68 missions from the 1972 NASA Payload Data Book and 9 from the 1971 Data Book. A total of 24 payloads was eliminated from the study, since they involved revisits and Sortie missions in which attitude control is supplied by the Orbiter or special payloads, such as large space stations and planet rovers, which are not compatible with the common hardware concept. The information given in the Payload Data Books was incomplete for many of the payloads. In areas where data was lacking, estimates were developed based on mission descriptions.

Initially, the payloads were tabulated in three-axis and spinning groups. These were in the order of payload number and included parameters affecting stabilization requirements such as pointing and attitude determination accuracies, reference orientation, altitude, payload weight, life, maintenance policy (expendable, on-orbit, or ground-refurbishable), and Shuttle visit intervals. In addition, the types of attitude control assemblies specified in the Payload Data Book were listed.

The kinds of stabilization sensors and actuators to be employed depend upon both the vehicle configuration and the type of mission. Possible vehicle types and missions were envisioned as follows:

Vehicle Configuration

Type 1: Three-Axis Pneumatic Control

Type 2: Yaw Spin (vehicle angular momentum cancelled by counter-rotating wheel)

Type 3: Three-Axis Momentum Control (control moment gyro or reaction wheel)

Type 4: Pure Spin

Type 5: Dual Spin

Missions

Class I: Earth Orbit, Earth Pointing

Class II: Stellar/Sun Orientation, Non-Earth Pointing, Heliocentric, Planet Flyby

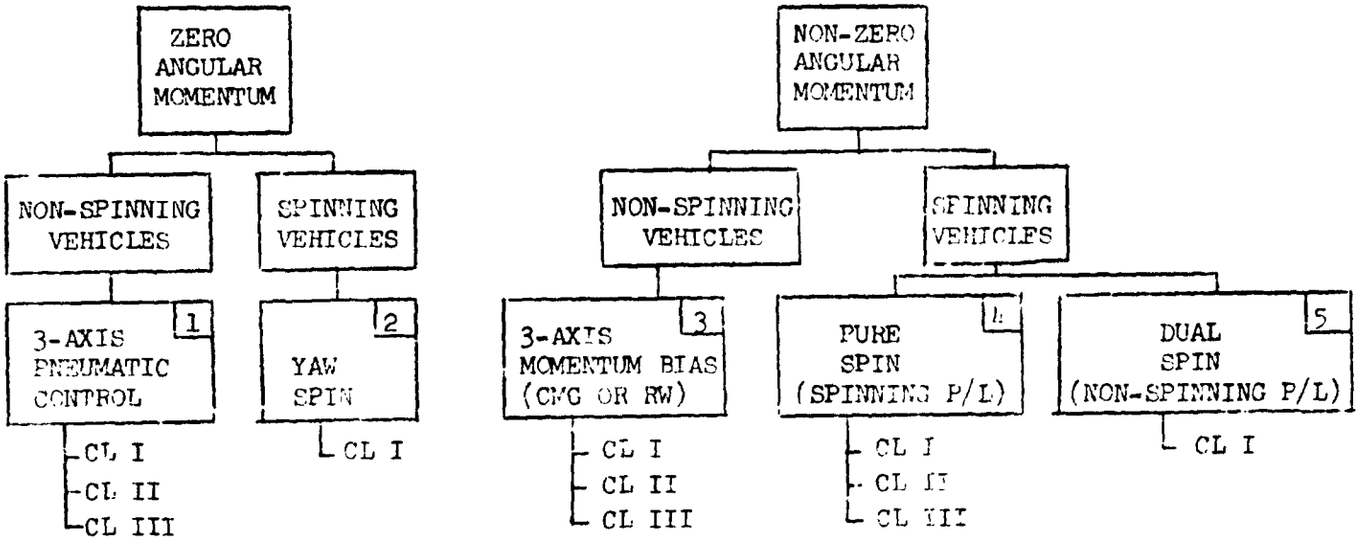
Class III: Planet Orbit

The vehicle types can be considered to fall into two major classifications: those having angular momentum[1] and those with zero angular momentun (see Figure 4-1). A review of the payloads covered by the study showed that Class I, II, and III missions were called for in Type 1, 3, and 4 vehicles. No missions involved Type 2, and all those under Type 5 were Class I.

The first grouping of payloads for determining common hardware potential was based on the premise that for a given vehicle configuration and mission class, the attitude sensor characteristics are largely determined by pointing accuracy, and the actuator ("muscle") requirements are dictated by pointing accuracy and vehicle weight. Therefore, all missions for each combination of vehicle configuration and mission class were arranged in order of decreasing pointing accuracy. Where accuracies were the same, the payloads were tabulated in order of increasing vehicle weight. A typical listing of this kind is shown in Table 4-2, which covers Type 3 vehicles (three-axis momentum control) and Class I missions (earth orbit, earth pointing). The horizontal lines show how the subsystems fall into four groups: (1) high accuracy, (2) medium accuracy/light weight, (3) medium accuracy/heavy weight, and (4) low accuracy. Furthermore, there could be considerable commonality of actuators and sensors. One actuator design would probably be required for the 136-kilogram (300-pound), lightweight vehicles, but, conceivably, a single additional design could handle all other vehicles, which range in weight from 454 to 1,361 kilograms (1,000 to 3,000 pounds). Common sensors could probably be used on both lightweight and heavyweight, moderate-accuracy systems.

[1] Angular momentum as used in this study does not refer to inertial angular momentum but is measured relative to a reference coordinate frame; e.g., a local vertical orbit plane reference frame.

REPRODUCIBILITY OF THE
ORIGINAL PAGE IS POOR



CLASS I EARTH ORBIT, EARTH POINTING
CLASS II STELLAR/SUN ORIENTATION (NON-EARTH POINTING, HELIOCENTRIC, PLANET FLYBY)
CLASS III PLANET ORBIT

NOTE: TYPES 1, 3, AND 5 ALL HAVE NON-SPINNING PAYLOADS, WHICH IMPLIES POTENTIAL HARDWARE COMMONALITY.

Figure 4-1. Classification of Attitude Control Subsystems

Table 4-2. Type 3 Vehicles, Class I Missions

PAYLOAD NO.	MISSION	REQUIREMENTS										
		NO. FLIGHTS	POINTING ACCUR. (MRAD)	ATTITUDE DETERM. (MRAD)	ATTITUDE (KM)	PAYLOAD WT. (KG)	GUID., NAV. & STAB. WT. (KG)	MASS EXPU. WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DUR (YEAR)	MAINTENANCE POLICY	SERVICE INTERVAL
NEO-11	Earth Resources	4	0.05	NG	SYN	454	29	54	6	3	R	NG
NC2-46	Meteorological	6	0.52	NG	SYN	1,361	136	94	5	3	O, R	NG
NEO-16	Meteor. & Resources	4	1.22	NG	926	1,134	91	109	12	2	O, R	1
NC2-47	S/C R&D	12	1.75	NG	SYN	136	9.1	16	1	1	O, R	1
NC2-48	S/C R&D	12	1.75	NG	556-5,556	136	9.1	16	1	1	O, R	1
NC2-50	Disaster Comm.	1	1.75	NG	SYN	798	54	82	15	5	O, R	NG
NE2-39	Earth Resources	5	1.75	NG	SYN	1,134	29	54	NG	2	O, R	NG
NE2-43	Earth Resources	1	1.75	NG	SYN	1,197	68	54	2	2	O R	NG
NE2-51	Traffic Cont. etc.	8	1.75	NG	SYN	1,297	156	141	5	5	O, R	NG
NE2-40	Meteorological	1	2.62	2.62	1,685	626	27	64	2	2	O, R	NG
NE2-42	Earth Resources	8	2.97	NG	926	816	68	27	2	2	R	2
NE2-38	Earth Resources	7	2.97	NG	926	1,089	68	27	NG	2	O, R	1
NEO-7	Cloud Observation	3	17.45	NG	1,296	454	27	50	12	4	R	NG

Abbreviations: NG - Not Given
O - On-Orbit Repair
R - Retrieve and Refurbish
SYN - Synchronous Orbit 34,800 kilometer (19,323 NM)

It appears desirable that certain other payloads having similar attitude control requirements should employ different configurations of stabilization hardware:

a. The mission might specify use of a slightly modified version of an existing spacecraft, such as Mariner, Pioneer, or Tiros. This would avoid the cost and risk involved with a completely new vehicle design.

b. A spinning payload might be needed for certain sensors used on the mission.

c. Large vehicles requiring frequent repointing should probably employ control moment gyros.

d. Certain types of missions may require some sort of momentum control in order to conserve propellant.

With these constraints in mind, the initial groups of the types shown in Table 4-2 were reduced in number by combining payloads that had similar attitude control requirements but with different stabilization hardware configurations specified in the Payload Data Books. This resulted in a total of 14 groups. Because of the unique requirements of certain missions, six of the groups consisted of only one or two payloads. The remainder of the groups had 3 to 13 missions. There was a total of 53 different payloads involving 177 flights. (See Tables 4-3 through 4-16.)

In the Payload Data Books, five of the satellites are specified to be the dual spin type. These are payloads NE2-41, NC2-49, NCN-7, NCN-8, and NCN-9. Since stabilization sensors and actuators are mounted on the non-spinning portion of the spacecraft, as on three-axis subsystems, it appeared that further hardware commonality could be achieved by changing the dual-spin types to the three-axis momentum bias configuration. Sensor and actuator requirements were similar to those on a group of three-axis subsystems employing the momentum bias technique, and the five dual spinners were shifted to that type of attitude control.

As mentioned above, six of the groups contain only one or two payloads, due to unique requirements. The payloads involved are as follows:

Table 4-3. Group 1

ATTITUDE CONTROL TYPE: 3-Axis

SENSORS: Rate Integrating Gyros, Sun Sensor, Star Sensor, Vidicon

ACTUATORS: Mass Expulsion

ACCURACY (MRAD): 0.297 PAYLOAD WT. (KG): 1,448

PAYLOAD NUMBER	MISSIONS	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ATTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DUR. (YEAR)
NU2-36	Encke Rendezvous	2	Sun	Encke	0.297	3 AU	1,448	3	3

TOTAL FLIGHTS: 2

Table 4-4. Group 2

ATTITUDE CONTROL TYPE: 3-Axis

SENSORS: Rate Integrating Gyros, Sun Sensor, Star Sensor,

ACTUATORS: Mass Expulsion

ACCURACY (MRAD): 8.73 - 17.5 PAYLOAD WT. (KG): 944 - 3,398

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ATTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DUR. (YEAR)
NU2-22	Mars Orbiter & Lander	2	Mars	Mars	17.5	1,502-33,036	3,398	2.5	2
NU2-25	Venus Orbiter	2	Venus	Venus	8.73	500	944	2	2
NU2-27	Mercury Orbiter	2	Merc.	Merc.	17.5	500	2,343	2	2
NU2-32	Jupiter Orbiter		Jupiter	Jupiter	17.5	1.1-1.4 R_j	1,134	3	3
NU2-34	Saturn Orbiter	2	Saturn	Saturn	17.5	1.2-2.3 R_s	1,074	6	6

TOTAL FLIGHTS: 10

Table 4-5. Group 3

ATTITUDE CONTROL TYPE: 3-Axis
SENSORS: Rate Integrating Gyros, Sun Sensor, Star Sensor
ACTUATORS: Mass Expulsion
ACCURACY (MRAD): 17.5 - 34.9 PAYLOAD WT. (KG): 530 - 2,263

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ATTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION (YEAR)
NU2-26	Venus Flyby	2	-	Space	34.9	-	530	0.5	0.5
NU2-29	Jupiter, Uranus Flyby	2	-	Space	17.5	-	699	6	6
NU2-33	Neptune, Uranus Flyby	2	-	Space	17.5	-	2,263	12	12
NU2-35	Encke Flyby	1	-	Space	17.5	-	1,433	2	2
NU2-37	Asteroid Rendezvous	2	SUN	Earth	17.5	3 AU	1,651	3	3

TOTAL FLIGHTS: 9

Table 4-6. Group 4

ATTITUDE CONTROL TYPE: 3-Axis
SENSORS: Earth Sensor, Sun Sensor
ACTUATORS: Mass Expulsion
ACCURACY (MRAD): 87.3 PAYLOAD WT. (KG): 168

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ATTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DUR. (YEAR)
NB2-55	Life Science	2	Earth	Earth	87.3	556	168	5	0.5

TOTAL FLIGHTS: 2

Table 4-7. Group 5

ATTITUDE CONTROL TYPE: 3-Axis

SENSORS: Special Two-Degree-of-Freedom Gyros, Earth Sensor, Sun Sensor, Star Sensor, Telescope

ACTUATORS: Reaction Wheels, Mass Expulsion

ACCURACY (MRAD): 0.0052 PAYLOAD WT. (KG): 349-463

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ALTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DUR. (YEAR)
NP2-16	Relativity Theory	2	Earth	Star	0.0052	926	463	1	1
NP2-17	Relativity Theory	2	Sun	Star	0.0052	0.3 AU-1.0 AU	349	1-2	1-2

TOTAL FLIGHTS: 4

Table 4-8. Group 6

ATTITUDE CONTROL TYPE: 3-Axis
SENSORS: Earth Sensor, Sun Sensor, Star Sensor
ACTUATORS: Momentum Bias, Mass Expulsion
ACCURACY (MRAD): 0.052 - 1.22 PAYLOAD WT. (KG): 322 - 1, 361

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ATTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DUR. (YEAR)
NE2-45	Earth Measurements	2	Earth	Earth	52.4*	500	322	12	3
NC2-46	Meteorological	6	Earth	Earth	0.52	SYN	1, 361	5	3
NEO-11	Earth Resources	4	Earth	Earth	0.052	SYN	454	6	3
NEO-15	Meteorological	2	Earth	Earth	0.052	SYN	454	12	2
NEO-16	Meteorological and Resources	4	Earth	Earth	1.22	926	1, 134	12	2

TOTAL FLIGHTS: 18

* Attitude Determination 0.140 mrad

Table 4-9. Group 7

ATTITUDE CONTROL TYPE: 3-Axis
SENSORS: Earth Sensor, Sun Sensor
ACTUATORS: Momentum Bias, Mass Expulsion
ACCURACY (MRAD): 1.75 - 8.73 PAYLOAD WT. (KG): 136 - 318

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ATTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DUR. (YEAR)
NE2-41	Cloud Observation	2	Earth	Earth	1.75	SYN	243	4	4
NC2-47	Spacecraft Subsystem R&D	12	Earth	Earth	1.75	SYN	136	1	1
NC2-48	Spacecraft Subsystem R&D	12	Earth	Earth	1.75	556-5,556	136	1	1
NCN-10A	Data Collection and Navigation	1	Earth	Earth	8.73	SYN	318	12	5
NCN-10B	Data Collection and Navigation	4	Earth	Earth	8.73	29,632-55,560	318	12	5

TOTAL FLIGHTS: 31

Table 4-10. Group 8

ATTITUDE CONTROL TYPE: 3-Axis

SENSORS: Earth Sensor, Sun Sensor

ACTUATORS: Momentum Bias, Mass Expulsion

ACCURACY (MRAD): 1.75 - 34.9 PAYLOAD WT. (KG): 454 - 1,554

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ATTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DUR. (YEAR)
NP2-13	Upper Atmosphere	6	Earth	Earth	34.9	185-3704	526	3	1
NE2-38	Earth Resources	7	Earth	Earth	2.97	926	1,089	NG	2
NE2-39	Earth Resources	5	Earth	Earth	1.75	SYN	1,134	NG	2
NE2-40	Meteorological	1	Earth	Earth	2.62	1,678	626	2	2
NE2-42	Earth Resources	8	Earth	Earth	2.97	926	816	2	2
NE2-43	Earth Resources	1	Earth	Earth	1.75	SYN	1,197	2	2
NC2-49	Satellite Commun.	6	Earth	Earth	1.75	SYN	798	12	5
NC2-50	Disaster Commun.	1	Earth	Earth	1.75	SYN	798	15	5
NC2-51	Traffic Control, Etc.	8	Earth	Earth	1.75	SYN	1,297	5	5
NCN-7	Commun. and Navigation	3	Earth	Earth	3.49	SYN	644	12	5
NCN-8	Commun. and Relay	3	Earth	Earth	3.49	SYN	1,554	12	7
NCN-9	Communication	2	Earth	Earth	3.49	SYN	454	11	5
NEO-7	Cloud Observ.	3	Earth	Earth	1.75	1,296	454	12	4

TOTAL FLIGHTS: 54

Table 4-11. Group 9

ATTITUDE CONTROL TYPE: 3-Axis

SENSORS: Rate Integrating Gyros, Sun Sensors, Star Sensors, Mission Equipment

ACTUATORS: Control Moment Gyros, Mass Expulsion

ACCURACY (MRAD): 0.0052 - 0.297 PAYLOAD WT. (KG): 7,158 - 14,643

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ALTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DURATION (YR.)
NA2-3	X-Rays	4	Earth	Star	0.297	463	7,531	4	2
NA2-5	Space Telescope	3	Earth	Star	0.0052	611	8,428	5	2
NA2-7	Solar Observation	1	Earth	Sun	0.052*	500	14,643	10	1
NA2-9	Celestrial Radiation	1	Earth	Star	0.0052	740	7,158	10	1

TOTAL FLIGHTS: 9

* Attitude Determination 0.0052 MRAD.

Table 4-12. Group 10

ATTITUDE CONTROL TYPE: Pure Spin
SENSORS: Sun Sensor, Star Sensor
ACTUATORS: Mass Expulsion
ACCURACY (MRAD): 0.052 PAYLOAD WT. (KG): 169

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ALTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DURATION (YR.)
NA2-1	Celestrial Radiation	12	Earth	Space	0.052	550	169	3	3
NA2-2	Celestrial Radiation	2	Earth	Space	0.052	Sync.	169	3	3

TOTAL FLIGHTS: 14

Table 4-13. Group 11

ATTITUDE CONTROL TYPE: Pure Spin
SENSORS: Sun Sensor, Star Sensor
ACTUATORS: Mass Expulsion
ACCURACY (MRAD): 17.5 - 34.9 PAYLOAD WEIGHT (KG): 259 - 3,946

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ALTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DURATION (YR.)
NP2-14	Ionospheric Measurements	6	Earth	Space	34.9	1,852-37,040	259	3	1
NP2-18	Magnetosphere	2	Earth	Space	17.5	12,779	1,973	6	3
NP2-19	Van Allen	2	Earth	Space	17.5	12,779	3,946	6	3

TOTAL FLIGHTS: 10

Table 4-14. Group 12

ATTITUDE CONTROL TYPE: Pure Spin
SENSORS: Sun Sensor, Star Sensor
ACTUATORS: Mass Expulsion
ACCURACY (MRAD): 17.5 PAYLOAD WT. (KG): 1,082

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ALTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DURATION (YR.)
NA2-11	Large Radio Antenna	1	Earth	Space	17.5	71,572	1,082	3	3

TOTAL FLIGHTS: 1

Table 4-15. Group 13

ATTITUDE CONTROL TYPE: Pure Spin
SENSORS: Sun Sensor, Star Sensor
ACTUATORS: Single Reaction Wheel, Mass Expulsion
ACCURACY (MRAD): 17.5 PAYLOAD WT. (KG): 290

PAYLOAD NUMBER	MISSION	REQUIREMENT							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ALTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DURATION (YR.)
NP2-15	Solar Wind	6	Sun	Space	17.5	1 AU	290	3	3

TOTAL FLIGHTS: 6

Table 4-16. Group 14

ATTITUDE CONTROL TYPE: Pure Spin

SENSORS: Sun Sensor, Star Sensor, Radio Signal Pattern

ACTUATORS: Mass Expulsion

ACCURACY (MRAD): 17.5 PAYLOAD WT. (KG): 279 - 884

PAYLOAD NUMBER	MISSION	REQUIREMENTS							
		NO. FLIGHTS	ORBIT	REFERENCE ORIENTATION	POINTING ACCURACY (MRAD)	ALTITUDE (KM)	PAYLOAD WT. (KG)	EXPECTED LIFE (YEAR)	MEAN MISSION DURATION (YR.)
NP2-20	Solar System Exploration	1	–	Earth	17.5	–	279	12	7
NU2-24	Venus Probe	1	–	Earth	17.5	–	398	1	1
NU2-28	Jupiter Orbiter	1	Jupiter	Earth	17.5	$1.1R_j$	884	4	4
NU2-30	Jupiter Flyby	2	–	Earth	17.5	–	360	3	3
NU2-31	Saturn Flyby	2	–	Earth	17.5	–	386	4	4

TOTAL FLIGHTS: 7

a. Explorer (NA2-1, NA2-2) - No other mission requires this combination of high pointing accuracy (~ 5 milliradians) and light vehicle weight (169 kg).

b. Radio Astronomy Observatory (NA2-11) - This is a spinning vehicle that stabilizes a circular antenna (10 kilometers in diameter) as well as four small outer satellites attached to the antenna. A special actuator design is expected to be required, since unusual dynamic problems appear likely.

c. Explorer (NP2-15) - This is a special spinning vehicle employing a single reaction wheel for varying the spin speed.

d. Gravity and Relativity (NP2-16, NP2-17) - These are relativity theory experiments that require very special gyroscopes with drift uncertainties of 7.69×10^{-17} radians/sec (0.0005 arc seconds per year), which is far beyond the state of the art.

e. Encke Rendezvous (NU2-36) - This comet randezvous mission requires a special vidicon sensor to achieve the specified pointing accuracy of 0.29 milliradians (1 arc minute).

f. Bio-Research Module (NB2-55) - The low accuracy pointing requirement (87.3 milliradians) can probably be achieved without the use of gyros or other momentum devices employed on all other three-axis vehicles.

Gravity gradient stabilization was specified in the Payload Data Book for Geopause (NE2-45) which requires a pointing accuracy of 52 milliradians (3 degrees). However, an attitude determination accuracy of 0.15 milliradians (30 arc seconds) is also specified, which will necessitate a star sensor. By changing from gravity-gradient to momentum bias stabilization, it was possible to group this payload with others of similar size which also have star sensors of the same order of accuracy.

Because magnetic torquing is feasible only over a limited range of altitudes, it has not been specified on any of the subsystems, since it would increase the number of groups, and this would be undesirable from the commonality standpoint.

2. RESULTS

Tables 4-3 through 4-16 divide the 53 payloads covered by this study into 14 groups. On the basis of the information contained in the 1971 and 1972 Payload Data Books, it appears that all of the missions tabulated in a given group could be handled by a common attitude control subsystem. Each table lists the system requirements and the types of sensors and actuators proposed for the common stabilization system. It will be noted that the ranges of accuracies and vehicle weights on each chart are restricted, so that the same sensor and actuator can be employed for all missions in the group.

On some charts, the range of mean mission durations is rather large, particularly on distant planet flybys which, of course, are expendable and cannot be serviced at regular intervals. In order to achieve the desired life and reliability on extended missions, it will be necessary to employ redundant, dormant assemblies. However, it should be noted that this does not affect commonality of hardware, since identical assemblies can be involved.

The redundancy required for meeting expected life or mean mission duration specifications could be based upon the design life of current hardware. However, the extent to which redundancy must be employed for enhanced reliability is beyond the scope of this study. It involves factors for which information is not available, such as the required reliability for the entire spacecraft and the extent to which the failure rates of other subsystems contribute to the overall reliability. Therefore, the number of redundant assemblies needed has not been specified at this time.

In the case of the Large Solar Observatory (NA2-7), which weighs 14,643 kilograms (32,282 pounds), redundant control moment gyros of the type used on lighter vehicles in the group will be required in order to meet torquing requirements.

A complete attitude control subsystem involves a number of different assemblies: attitude sensors, sensor electronics, actuators, actuator electronics, interface electronics between the sensors and actuators,

and a computer for such functions as control law implementation, sequencing discretes, and failure detection analysis. For the concept of on-orbit servicing, subsystems will presumably be modularized. These modules must be accessible to the Orbiter or Tug for on-orbit checkout, replenishment of consumables, and for removal and replacement.

Since different functional considerations determine the location of sensor and actuator assemblies, it seems logical to assume that they will be in separate modules. Also, it appears desirable to include the associated electronics in each module. Different groups of missions will require different combinations of sensor and actuator modules. Therefore, for module commonality, the interface electronics should be in a separate module.

Off-the-shelf hardware could probably satisfy the requirements for most of the groups of common attitude control subsystems shown in Tables 4-3 through 4-16 Specific hardware has been identified in Table 4-17 for each of the 14 groups, where currently available sensor and actuator assemblies appear capable of meeting the performance requirements. This has been done to show the feasibility of employing one common assembly for handling the range of performance covered by all of the missions in each group. Actually, these particular pieces of hardware would probably not be employed. The missions involved are scheduled far into the future, and improved, more reliable equipment will undoubtedly be available by that time. Also, the concept of replaceable modules will almost certainly result in extensive design changes on current hardware. Table 4-18 summarizes the common hardware assemblies required for the 14 groups.

C. SOLAR ARRAY

Interest in reducing the cost of solar cell power systems in recent years has given rise to the concept of solar array standardization as a potential cost savings technique. It is reasonable to think that the present arrays which are custom designed specifically for each mission are more costly than a solar array design which has been standardized. An uncertainty which clouds the feasibility of standardization, however, is the possibility of increased array weight and array area which might ultimately result in

Table 4-17. Applicable Current Hardware

GROUP NO.	ASSEMBLY	MANU-FACTURER	MODEL	REMARKS
1	Rate-Integ. Gyros	Lockheed	ARA	
	Sun Sensor	ADCOLE	1401	
	Star Sensor	TRW	Pioneer F/G	
	Vidicon			Special Design
	Jet Thrusters	Lockheed	Agena	
2	Rate-Integ. Gyros	Lockheed	ARA	
	Sun Sensor	ADCOLE	1401	
	Star Sensor	TRW	Pioneer F/G	
	Planet Sensor			Data Unavailable
	Jet Thrusters	Lockheed	Agena	
3	Rate-Integ. Gyros	Lockheed	ARA	
	Sun Sensor	ADCOLE	1401	
	Star Sensor	TRW	Pioneer F/G	
	Jet Thrusters	Lockheed	Agena	
4	Earth Sensor	Barnes	13-181X	
	Sun Sensor	ADCOLE	1401	
	Jet Thrusters	Lockheed	Agena	
5	2-Deg-Free-Gyro			Special 7.69 X 10^{-14} mrad/sec
	Earth Sensor	TRW	AOGO	
	Star Sensor	ITT	OAO Boresighted	Telescope sensor for fine pointing
	Sun Sensor	Bendix	1818823	
	(3) Reaction Wheels	Sperry	45Q RWA	
	Jet Thrusters	Lockheed	Agena	
6	Earth Sensor	TRW	AOGO	
	Sun Sensor	ADCOLE	1401	
	Star Sensor	ITT	OAO Boresighted	
	Momentum Wheel	Sperry	45Q RWA	
	Jet Thrusters	Lockheed	Agena	

Table 4-17. Applicable Current Hardware (Continued)

GROUP NO.	ASSEMBLY	MANU-FACTURER	MODEL	REMARKS
7	Earth Sensor Sun Sensor Momentum Wheel Jet Thrusters	TRW ADCOLE Bendix Lockheed	AOGO 1401 1880272 Agena	
8	Earth Sensor Sun Sensor Momentum Wheel Jet Thrusters	TRW ADCOLE Sperry Lockheed	AOGO 1401 45Q RWA Agena	
9	Rate-Integ. Gyros Sun Sensor Star Sensor CMG's Jet Thrusters	Lockheed Bendix ITT Bendix Lockheed	ARA 1818823 AOA Boresighted HEAO Agena	Telescope sensor fine pointing
10	Sun Sensor Star Sensor Jet Thrusters	ADCOLE ITT Lockheed	1301 OAO Boresighted Agena	
11	Sun Sensor Star Sensor Jet Thrusters	ADCOLE TRW Lockheed	1301 Pioneer F/G Agena	
12	Sun Sensor Star Sensor Jet Thruster	ADCOLE TRW Lockheed	1301 Pioneer F/G Agena	Dynamic Probl. expected that may require special actuator design
13	Sun Sensor Star Sensor (1) Reaction Wheel Jet Thrusters	ADCOLE TRW Bendix Lockheed	1301 Pioneer F/G 1880272 Agena	On spin axis

Table 4- 17. Applicable Current Hardware (Continued)

GROUP NO.	ASSEMBLY	MANU-FACTURER	MODEL	REMARKS
14	Sun Sensor	ADCOLE	1301	
	Star Sensor	TRW	Pioneer F/G	
	Radio Signal Pattern			Pioneer Design
	Jet Thrusters	Lockheed	Agena	

Table 4-18. Common Hardware Assemblies for Stabilization Systems

ASSEMBLY	REPRESENTATIVE MANUFACTURER	MODEL	ACCURACY MILLIRADIANS (DEG)	COMMENTS
Rate-Integrated Gyros	Lockheed	ARA	1.7×10^{-3} mrad/sec (0.35° /hr)	3 SDF, Kearfott 2564 Gyros
Sun Sensor	ADCOLE	1401	4.4 (0.25)	3-Axis Vehicle
Sun Sensor	ADCOLE	1301	4.4 (0.25)	Spinning Vehicle
Sun Sensor	Bendix	1818823	0.024 (0.0014)	Null Seeker
Star Sensor	TRW	Pioneer F/G	8.7 (0.5)	Solid State Scan
Star Sensor	ITT	OAO Boresighted	(0.0013)	Electronic Scan
Earth Sensor	Barnes	13-181X	3 (2)	Radiation Balance
Earth Sensor	TRW	AOGO	1/0 (0.06)	2-Axis Sweep Through
Momentum Wheel	Sperry	45Q RWA	-	61 newton-meter-sec (45 lb-ft-sec)
Momentum Wheel	Bendix	1880272		2.8 newton-meter-sec (2.06 lb-ft-sec)
CMG	Bendix	HEAO	-	611 newton-meter-sec (450 lb-ft-sec)
Jet Thrusters	Lockheed	AGENA	-	0.23 kg to 4.5 kg (0.5 lb to 10 lb) thrust representative

an increase in total program costs. One recent study group (Ref. 13) concluded that standardization appears to be feasible in the Shuttle time frame when potential solar array design freedoms such as the relaxation of s ze and weight constraints become more practical.

This study attempts to make a first-order estimate of the cumulative solar array cost savings that may be gained by means of solar array standardization for Shuttle payloads. The weight, area, and cost of standardized and custom arrays are compared for several types of typical missions.

1. SIZING OF SOLAR ARRAYS

Table 4-19 summarizes the results of the solar array sizing analysis. Column 1 is a summary of the missions and types of solar arrays which are typical of earth-orbiting spacecraft using solar power systems. Average array temperature and charged particle radiation fluence given in columns 2 and 3, respectively, are the major orbit-dependent environmental factors that affect array performance. For the customized arrays, a 10 Ω-cm solar cell was selected for use for synchronous and 12-hour elliptical orbits because of the more severe radiation environments compared to a low earth orbit. The maximum power point of single cells at the end of a mission (column 4) was determined from the calculated I-V curves shown in Figures 4-2 and 4-3. The number of single solar cells connected in series (column 5) required for the array to operate at the maximum powerpoint is based on an array operating voltage of 28 volts. The area of each customized array is based on a nominal array output of 1 kw. This area scales linearly with power if array size at other power levels is desired.

The standardized array must be capable of meeting the requirements of several types of missions, even though some loss of performance would be expected. The series configuration of 71 cells was selected as the reference configuration for the standard module because this configuration would meet the performance requirements of all the missions considered. The size and weight impact of using the standard module was determined by calculating the I-V curve operating point for each mission, followed by the

Table 4-19. Sizing Results for 1 KW Solar Array

Mission	1	2	3	4	5	6	7	8	9
	Ave. Array Temp (°C)	Radiation* Fluence (1 mev e/cm^2)	Cell Resistivity (ohm-cm)	Max Power** (Watts/ Cell)	No. Cells¥ in Series Custom Array	Array Area, Custom *** (m^2)	Array Area, Stand. **** (m^2)	Δ Area Col. 7- Col. 6 (m^2)	Δ Weight (kg)
Synchronous, Body Mounted, Spinning	15								
5 Years		3×10^{14}	10	0.049	67	29.2	31.0	1.86	6.4
10 Years		6×10^{14}	10	0.044	69	33.7	34.1	0.37	1.4
Synchronous Oriented Panels	60								
5 Years		3.5×10^{14}	10	0.045	68	9.3	9.5	0.19	5.9
10 Years		7×10^{14}	10	0.047	70	9.9	10.2	0.28	2.3
Low Earth, 3 Years									
Body Mounted, Spinning	20	1×10^{14}	2	0.045	63	33.2	35.3	2.14	7.3
Oriented Panels	54	1.2×10^{14}	2	0.046	63	10.6	11.0	0.37	2.7
12-Hr Elliptical, Oriented Panels	10								
3 Years		6×10^{14}	10	0.044	70	11.0	11.2	0.28	2.3
5 Years		10×10^{14}	10	0.041	71	11.5	11.5	0	0

* Cells covered with 12 mil coverslides
** Based on 2 x 2 cm cell
*** Based on 1 KW output
**** Based on a standard module, 10 ohm-cm cell, 2 x 2 cm, 71 connected in series, 1 KW output
¥ Number required to operate at maximum power at 28 VDC

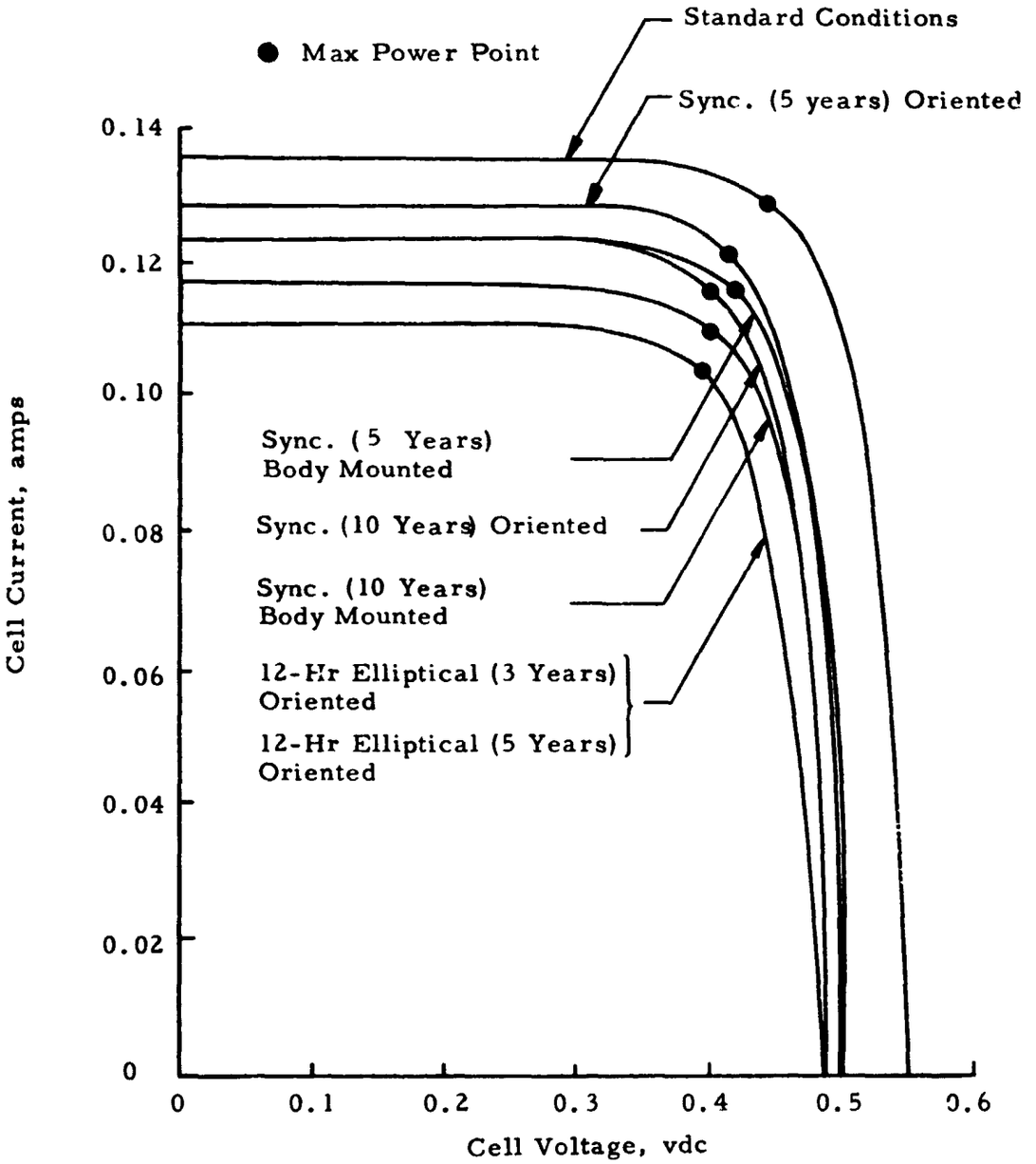


Figure 4-2. Solar Cell I-V Curves
(10 Ω-cm, 2 × 2 cm)

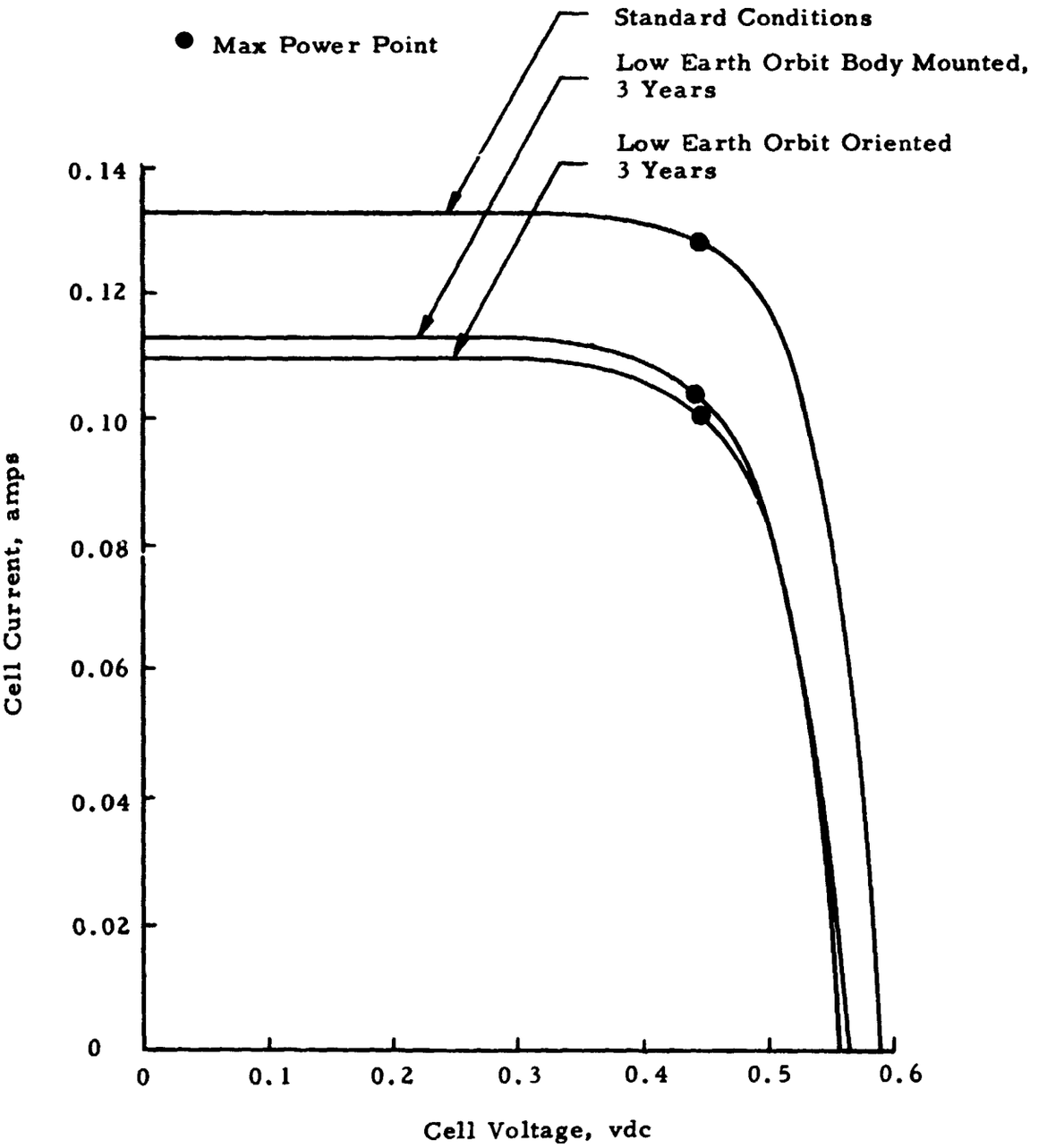


Figure 4-3. 1 Ω-CM Solar Cell I-V Curves

calculation of an array area for the mission. In estimating the weight impact, a value of 7.3 kg/m^2 (1.5 lb/ft^2) was used for both customized and standardized oriented arrays and is consistent with use of the Shuttle where array weight is not expected to be a major design constraint. For body-mounted arrays, it was assumed that 3.4 kg/m^2 (0.7 lb/ft^2) of substrate and structure were chargeable to the solar array.

Table 4-20 lists those general characteristics of a standardized array which seem likely at this time. It is expected that relatively thick and large-area solar cells would be used to minimize solar cell and array fabrication cost. It is also expected that standard array modules would not be designed to maximize array power-to-weight ratio and, as such, would be constructed using relatively heavy substrate. The weight penalty incurred by the use of thick cells is partially compensated by the increase in cell efficiency with cell thickness (Ref. 14). Also, large area cells increase the array packing factor, which is defined as the ratio of active array area to total array area. Array structural design, however, also depends on thermal design and stiffness requirements. Further construction and material details, therefore, cannot be identified until standard cell and standard array specifications are developed which define these requirements. Table 4-20 also shows typical values for design parameters for custom arrays which been flown.

2. POTENTIAL COST SAVINGS

The estimate presented herein of the potential cost saving realized from array standardization in the 1979-1990 time frame is a first-order approximation based on certain key assumptions concerning future solar array utilization in space and solar cell cost projections. In general, solar array cost can be subdivided into recurring and nonrecurring costs. Recurring costs include costs of materials (solar cells, covers, etc.), fabrication labor, management support, and acceptance testing. Nonrecurring costs include design analysis, prototype and qualification testing, labor, and test equipment. Recurring costs tend to vary directly with the amount of flight hardware built, whereas nonrecurring costs, as a first approximation, are independent

Table 4-20. General Characteristics of Custom and Standard Solar Array Modules

COMPONENT	CUSTOM MODULE (TYPICAL)	STANDARD MODULE
SOLAR CELL		
Type	Silicon N/P	Silicon N/P
Size	2 x 2 cm, 2 x 4 cm, 2 x 6 cm	2 x 6 cm
Thickness	6 - 14 mil	12 - 14 mil
Resistivity	1 - 10 Ω - cm	10 Ω - cm
COVERSLIDE		
Type	Fused Silica or Microsheet	Fused Silica
Thickness	6 - 40 mil	10 - 12 mil
SUBSTRATE	Al Honeycomb or Kapton	Al Honeycomb

of the amount of flight hardware built. A previous study (Ref. 15) has shown that array recurring costs tend to correlate relatively well with array area as opposed to maximum or average array power output. This correlation is used in the present analysis.

The mission planning model summarized in Table 4-21 was used to estimate the cumulative solar array capacity required for DOD, NASA, and other Shuttle payloads over the 12-year period, 1979-1990. Using the results of the previous subsection, the average yearly production requirement for custom versus standard array panels is 622 m^2/year (6700 ft^2/year) and 678 m^2/year (7300 ft^2 year), respectively.

The data shown in Figure 4-4 (Ref. 15) were used to estimate solar array recurring cost per unit array area. The recurring costs include the cost of materials, fabrication labor, test, and management and support. These data indicate that array costs per unit array area are reduced as larger units are built. For custom arrays, it was assumed that the array for each of the 42 spacecraft launched per year were procured individually, so that the average procurement is 14.9 m^2/spacecraft (160 ft^2/spacecraft). This assumption tends to maximize the cost per unit area, because some projects would involve multiple launches of identical array designs, and the average procurement would exceed 14.9 $meter^2$ (160 ft^2). The average cost for custom arrays is \$75,347/m^2 (\$7000/ft^2), so that the total array recurring cost for the 1979-1990 time frame is approximately \$570 million.

The concept of a standard array module permits long-term purchasing and the stockpiling of modules by users or manufacturers. Under these conditions, the size of a single procurement by the user or a centralized agency would exceed that needed for a single flight project. It is assumed here that NASA and DOD needs are equivalent and that each obtains 339 m^2/year (3650 ft^2/year) of standard array modules as a single procurement. Under these conditions, Figure 4-4 shows that the recurring cost of standard modules is \$32,300/m^2 (\$3000/ft^2), so the total recurring cost for the 500 spacecraft is \$260 million.

Table 4-21. Solar Array Planning Model

Time Period: 1979 - 1990	
Type of Solar Array by Orbit	**Relative Number of Launches**
Synchronous	
Body Mounted	25%
Oriented Panel	19%
Low Earth Orbiting	
Body Mounted	19%
Oriented Panel	25%
Elliptical	
Oriented Panel	12%
Average Array Area per kw[1], Customized: 19.1 m^2/kw (205 ft^2/kw)	
Average Array Area per kw[2], Standardized: 20.8 m^2/kw (224 ft^2/kw)	
Total Number of Spacecraft: ~ 500	
Average Array Power Output per Spacecraft: ~780 watts	
Solar Array Requirements Customized Modules: 7,430 m^2 (80,000 ft^2) Standard Modules: 8,080 m^2 (87,000 ft^2) Cumulative Weight Penalty: 4.76 x 10^3kg (10.5 x 10^3 lb)	

1. Using Table 4-19, Column 6 data, the percentages above.
2. Using Table 4-19, Column 7 data, and the percentages above.

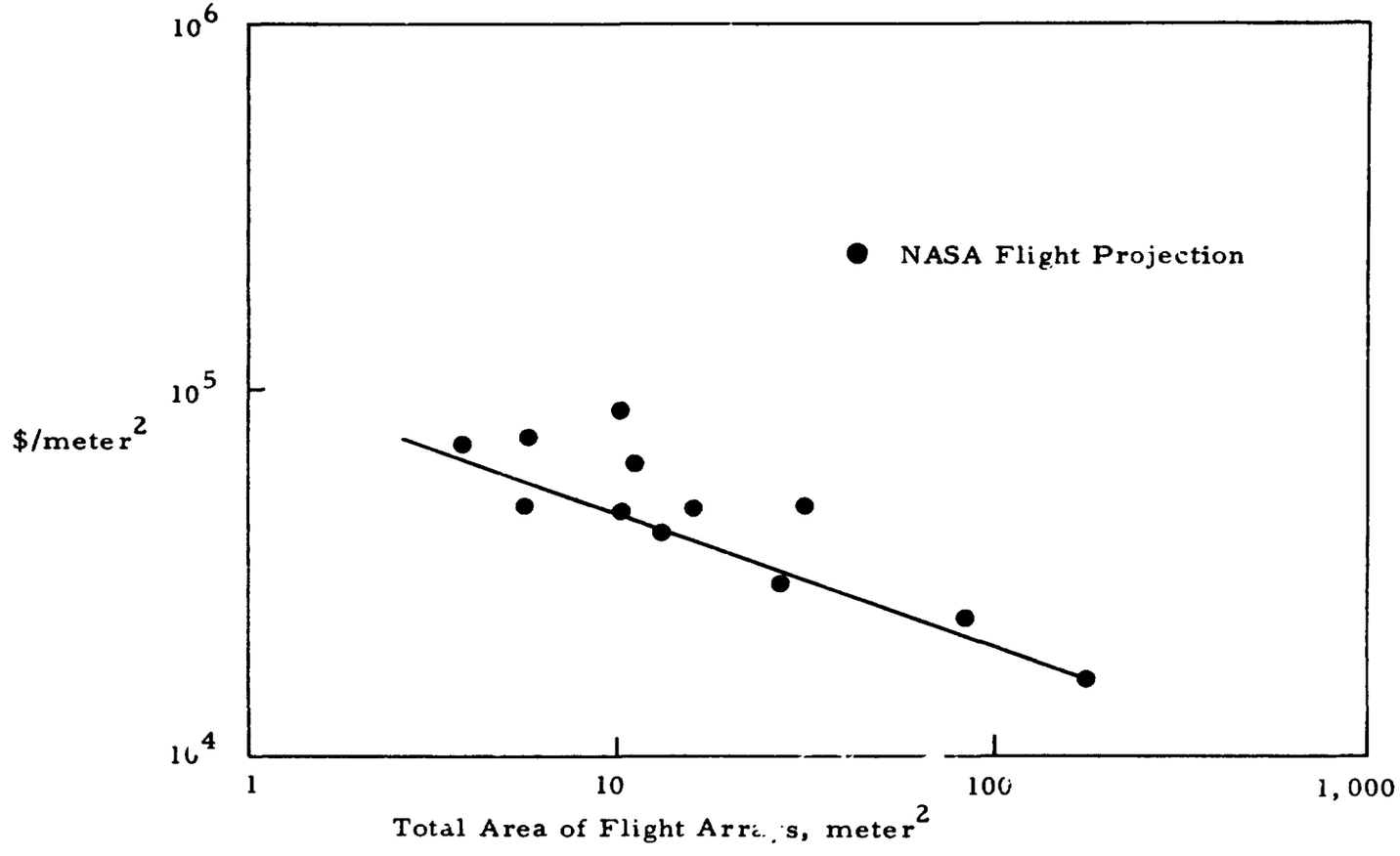


Figure 4-4. Solar Array Recurring Cost

Additional savings in solar array cost can be realized because of the expected reduction in solar cell costs, as indicated by the curves shown in Figure 4-5. The savings are a result of the so-called "learning effect" where costs are reduced as the cell manufacturer gains experience in repetitive manufacturing operations. The cost of standard cells would probably show a faster initial decline, since the standard cell would be the same as or very similar to a cell already in production and only cells of a single type and size would be produced. Using the Figure 4-5 data, it is estimated that custom cell costs would decrease an average of 14 percent during the period 1970-1990, whereas the cost of a standard cell is estimated to decrease an average of 25 percent. The extrapolated portions of the curves in Figure 4-5 are based on the assumption that no new solar cell technology is introduced in the interim period. The learning curve for the standard cells is expected to flatten out around 1990, because by then the technology and manufacturing methods for their production would have been well established. Since solar cell costs are approximately 30 percent of panel cost, the average expected reduction of cost for custom and standard flight panels through learning is 4 percent and 7 percent, respectively. Applying these percentage reductions to the recurring cost estimates developed above gives total recurring costs for custom and standard arrays of $550 million and $240 million, respectively.

Potential cost reductions of other array components resulting from the learning effect have been neglected. This is justified on the basis that the production technology for array substrates and other electrical components is well established, and any potential savings can be neglected compared to the potential solar cell cost savings.

Estimates of nonrecurring costs for customized and standard array modules are much more difficult because of the general lack of detailed historical cost data. A recent study (Ref. 16) has shown that nonrecurring costs range from 0 percent to 50 percent of total array cost and depend, to a large extent, on program requirements rather than technology requirements. Major nonrecurring cost items include those associated with array

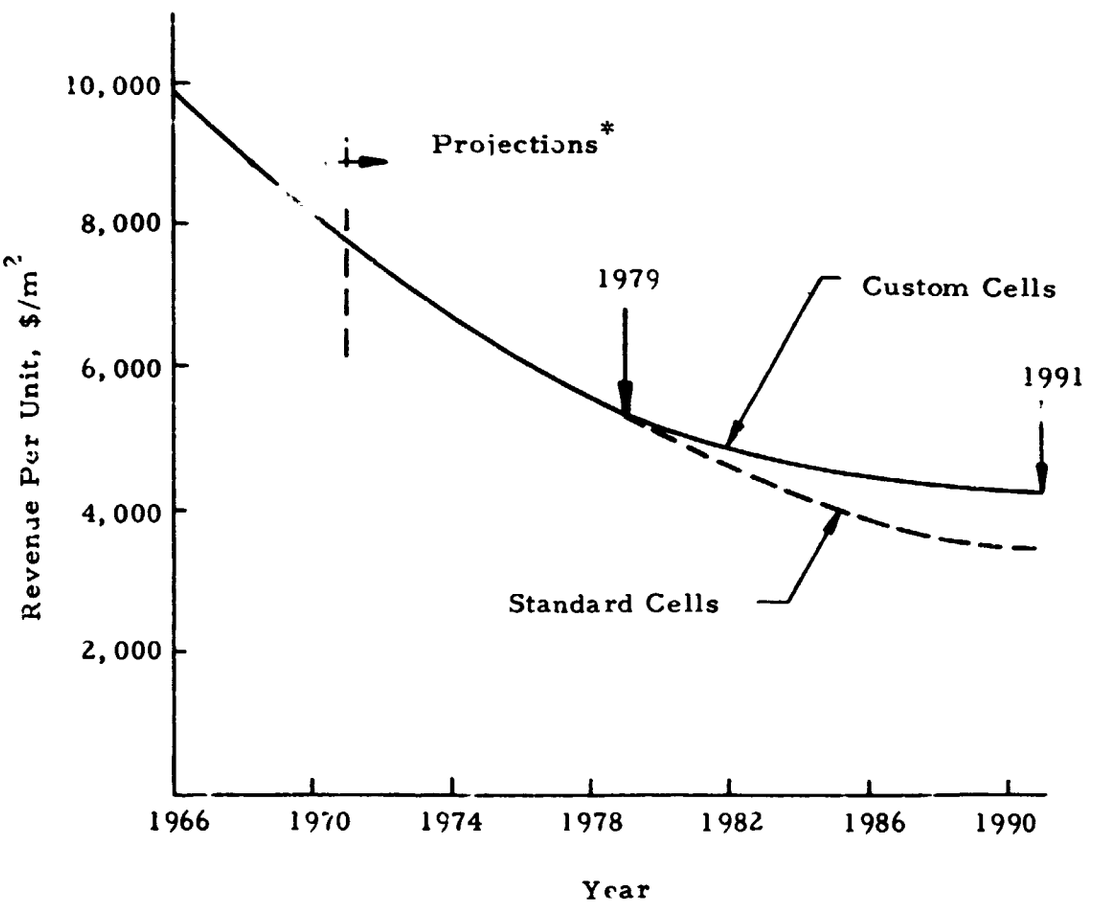


Figure 4-5. Silicon Solar Cells Learning Curve

*This figure was provided by E. L. Ralph of Spectrolab/Heliotek, a Division of Textron, who calculated the projections shown.

Table 4-22. Solar Panel Cost Breakdown

ITEM	RELATIVE COST, %
DESIGN ANALYSIS*	5.0
FABRICATION	
Labor	20.0
Material	27.0
Quality Control	7.0
Documentation	6.0
Management & Support	13.0
PANEL TEST	
Prototype*	0.9
Qualification Test*	0.9
Acceptance Test	11.0
Quality Control	5.5
Test Equipment*	3.7

* Nonrecurring cost items

design and analysis, prototype fabrication and testing, and test equipment. The potential savings that may be realized via standardization are even more uncertain. Table 4-21 shows relative array cost items for a current NASA program (Ref. 17). These data show that approximately 10 percent of the total array cost can be considered as nonrecurring. In the present analysis, it is assumed that the nonrecurring costs during the period of interest are approximately 20 percent of the total cost of customized arrays. For standard arrays, it is assumed that nonrecurring costs constitute only five percent of total array costs. Based on these assumptions, the estimated total nonrecurring costs for custom and standard arrays are approximately $60 million and 13 million, respectively. Total costs are summarized in Table 4-23.

Table 4-23. Estimate of Total Solar Array Cost, 1979 - 1990

Item	Customized Array, $M	Standard Array, $M
Recurring Cost	550	240
Nonrecurring Cost	60	13
Total Cost	610	253
Net Savings		357

D. CONCLUSIONS

As anticipated, this study has indicated that commonality of attitude control subsystem hardware should be very advantageous for Shuttle payloads. Minimizing the number of different sensors and actuators should result in large cost savings in engineering, manufacturing, test, maintenance, and repair. In addition, it should provide more reliable equipment having significant performance margins.

Common hardware should be employed on payloads having similar requirements. Candidates for common hardware were selected by first grouping types of missions (earth orbiting and earth pointing, for example) and then establishing subgroups with similar accuracies and weights.

A total of 53 different payloads involving 177 flights was considered in the study. It appears that only 14 different types of attitude control subsystems could accommodate all of these payloads. In addition, considerable hardware commonality would occur among sensor and actuator modules.

The results of this study have also shown that the use of standard solar array modules during the 1979-1990 time period for several different types of mission projects could result in significant savings in solar array costs while incurring relatively minor array size and weight penalties. These results, however, hinge on several key assumptions which tend to maximize the cost of custom arrays and minimize the cost of standard arrays. These assumptions must be verified and modified, if necessary, in a more detailed analysis before a realistic appraisal of standardization can be made. The indicated cost saving potential of $357 million is great enough so that a more thorough investigation seems warranted.

E. RECOMMENDATIONS

Hardware commonality should be a basic policy for attitude control subsystems on Shuttle payloads. Design studies should be initiated on a minimum number of different sensors, actuators, and interface electronics modules suitable for all anticipated Shuttle missions. In view of the cost savings which result from hardware commonality, it is recommended that the subsystem modules be identical for expendable, on-orbit repair, or retrieve/refurbish missions. An additional reason is that all subsystems, even those on expendable missions, should be checked out by the Shuttle or Tug when the payload is placed in its final orbit or in a parking orbit in order to insure that no malfunctions occurred during ascent. Commonality of spacecraft hardware will result in standardized checkout test equipment and procedures aboard the Shuttle or Tug and will permit replacement of any failed modules without bringing the spacecraft back to earth.

Based on the results of the present study, it is recommended that the feasibility and potential advantages of solar array standardization be investigated in more detail. Using the present results as a point of departure, this follow-on work should include the following:

1. The feasibility of a specification for standard cells and/or modules capable of meeting the requirements of a wide variety of NASA and DOD missions

2. The impact of array hardening requirements on the feasibility of standardization

3. A detailed estimate of the potential nonrecurring cost savings resulting from array standardization

4. The effect of cell and/or array module standardization on the development and implementation of improved solar cells and module fabrication techniques

5. SORTIE PAYLOAD OPERATIONS

A. INTRODUCTION

It has been suggested that certain Sortie payloads in the NASA mission model may have substantially overlapping data requirements with other payloads and that a cost savings could be obtained by flying these pairs as joint payloads. One such pair is the combination of the Solar Shuttle Observatory (SSO) and the Atmospheric Science Facility (ASF) because the ASF requires substantial quantities of solar data for its own use.

The first phase was to determine the instrument complement for a joint mission, the requirements this complement would place upon an integration facility (or facilities), and the orbital operations this complement would demand of the astronaut and mission specialist. The joint mission could then be utilized to construct a general model of the requirements the scientific instrumentation would place on integration facilities, Sortie modules, and specialists.

The second phase examined the compatibility of the experiment equipment operations with the operational requirements of the Space Shuttle. This effort evaluated the experiment requirements, operational sequences, time lines, and facilities.

Since results from the first phase showed ASF-SSO is not a useful combination, a modified Atmospheric Science Facility was selected to provide a model for the integration facility impact study and the orbital operation study. The modified ASF described here is not a current NASA description.

The modified Atmospheric Science Facility is based upon the Martin Marietta Corporation Study of an Atmospheric Science Facility (Ref. 19).

Since the Martin study was published, there have been continuing formal and informal critiques. The modified ASF attempts to incorporate many of the ideas expressed in these critiques and in discussions with Dr. R. D. Hudson of the Johnson Space Center. Comments of the NASA working group review of the Martin study and a few ideas generated in the Space Physics Laboratory of The Aerospace Corporation have also been implemented. The resulting modified Atmospheric Science Facility is utilized to provide input data for the second phase of this study. The exact instrument complement is not vital to the overall study since the final models are supposed to be general guidelines for planning purposes. Consequently, the modified Atmospheric Science Facility has not been submitted to Dr. Hudson for explicit approval, though the concept has been approved for the purpose of this study. The modifications are made to utilize a payload which is somewhat more complex and therfore more like other payloads, involves man to a greater degree, and should be more nearly representative of the final concept of this evolving facility.

B. ANALYSIS OF A JOINT ASF/SSO PAYLOAD

The analysis of a joint payload concept for these two missions includes a determination of potentially common instrumentation, data-collecting techniques, and orbital requirements.

A summary of the ASF instrumentation is given in Table 5-1. This table includes an XUV-UV solar intensity monitor and a high-resolution solar emission line spectrograph.

Table 5-2 lists candidate instruments for inclusion on the SSO. This list is taken from Reference 19 and includes an 0.3-m telescope and associated spectroheliograph for the ultraviolet as well as a 0.5-m telescope and associated spectroheliograph for the extreme ultraviolet solar radiation. When the SSO instrumentation is casually inspected, it appears that all of the instrumentation would be useful to the ASF experimenter. However, the astronomer is interested in very small regions on the solar disc. If the

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Table 5-1. ASF Facility - Baseline of Martin Story

Experiments	Observation	Requirements: Interval	Requirements: Resolution	Time, Δt	fov	Dimensions (m): L	W	H	Weight (kg)	Remarks
XUV Normal Incidence Grille Spectrometer	XUV airglow spectrometer, light emitted and scattered	300-1300 Å	10 Å	1 sec	10° × 10°	0.76	0.25	0.1	11.4	0.1 R at SR4 sensitivity, internal calibrate, 5% absolute intensity, 0.1-200 R dynamic range
UV-Visible - NIR Spectrometer	Atmospheric emission and scattered light detector for excited states, aerosol scattering, etc.,	0.1150-1 μm	0.1 Å, λ < 3000 Å	1 sec	0.3 × 6 min and 0.7 × 2.5°	1.1	0.56	0.38	86.5	0.1 R sensitivity ±10% relative intensity, 10^5 dynamic range
High-Resolution Infrared Interferometer Michelson Cryo-cooled LWIR	High-resolution spectrometric data in the IR to study minor constituents, reactions, etc.	1-5 μm	0.05 cm^{-1}	3 min	Selectable 3, 5, 15, 30 min 1.5°	0.72	0.3	0.52	45	10^{-11} W/cm^2 sr-μm sensitivity 10^6 dynamic range
		5-150 μm	0.1 cm^{-1}	3 min		1	0.75	0.75	110	
XUV-UV Solar Intensity Monitor	Precise monitoring of total solar energy for earth's radiation budget	0.0180 - 0.170 μm	2 Å	0.25 sec	7 × 32 min					<1% absolute intensities in flight calibration, dynamic range ≈ 5 of 10^4
High Spectral Resolution Line Profile Spectrograph	Determine line shapes (i.e., T and velocity fields) of prominent solar UV emission lines	3 regions independently 300-600 Å 580-1220 1150-1800	0.02 Å	<5 min	Optical system limits - 32 min by TBD	1.7	0.61	0.10	32	Grazing incidence, normal incidence, Echelle spectrograph
						1.4	0.46	0.31	57	
						0.86	0.3	0.3	27	
Filter Photometer	For previewing emission features that may take an unusual characteristic, for monitoring atmospheric scattering	Four sets of four each. 1. scattering 2. O_3 scatter, 3. O_2 scatter and 4. TBD	15 Å		1 min to 1° Selectable	0.76	0.2	0.2	4.5	Four instruments each physically alike but with different filters and detectors
						0.76	0.2	0.2	4.5	
						0.76	0.2	0.2	4.5	
						0.76	0.2	0.2	4.5	
UV Documentation Camera	To record the image of what the instruments see, i.e., aurora, red arcs, etc.	2400-7000 Å	≈250 Å	0.2 sec	30	0.3	0.3	0.3	2	Two Nikon cameras

Table 5-2. Candidate List for Instruments for Inclusion on a Solar Shuttle Observatory

Instrument	Aperture m	Wavelength Range	Resolution Spatial arcsec	Resolution Δλ Å	Resolution Δt secs	Field of View	Wt Kg	Provisional Dimensions, m W	Provisional Dimensions, m H	Provisional Dimensions, m l	Remarks
Photoheliograph	1.0	0.2-2.0 μm	0.15	.01	1	4 arcmin	2200	1.8	2.0	3.0	Primary image photograph, spectroscopy
Spectrograph/ spectroheliograph	-	0.2-1.5 μm	0.15			4 arcmin	500	0.5	1.0	3.0	Spectra, magnetic and velocity fields
Ultraviolet telescope	0.3	0.06-0.2 μm	0.3	.01	1	4 arcmin	400	1.3	1.5	3.0	Primary image photograph, spectroscopy
Ultraviolet spectroheliograph	-	0.06-0.2 μm	0.3			4 arcmin		0.5	0.5	3.0	Spectra, line profiles, photoelectron scan
Coronagraph	0.9	0.4-0.7	1.0	2000	10	15 deg	410	0.7	0.9	3.4	Corona, 1-6, 5-30 solar radii
Extreme ultraviolet spectroheliograph	0.5	100-600Å	1.0	.02	.2	5 arcmin	400	0.5	1.0	10.0	Spectroheliograms, line profiles
X-ray spectroheliograph	0.6	5-140 Å	1.0	.01	.2	5 arcmin	400	0.5	1.0	10	Spectroheliograms, filtergrams
Crystal spectroheliograph	0.25	1-6 Å	2.0	.001	.2	2 arcsec (rastered)	500	0.5	2.0	8.0	Spectral line profiles
Flare detector	0.1	3-25 Å	<1.0	-	.3	Full disk	100	0.1	0.1	10	Active region, flare location and study
Vector Magnetograph	-	.2-2	0.15	.1	.3	4 arcmin	30	0.3	0.3	0.5	Associated with telescope, item 1

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field-of-view (FOV) requirements on the SSO ultraviolet instrument are compared with the solar disc area, it can be seen that a raster scan and summation of ~64 FOV elements would be required to provide the solar disc total energy. The UV monitoring function or the ASF facility requires about 370-m (0.2-nmi) resolution, but the SSO would yield 1.9m (0.001 nmi). In order to obtain intensities over a full line width for ASF purposes, either the SSO instrument focusing properties must be degraded, thereby making the data worthless to the solar astronomer, or the data must be obtained by scanning and summing the full width of the line at each of the 65 FOV element locations. If the data were to be used by both facilities, the solar astronomer would not be able to continuously monitor a small active region on the sun, and the ASF experimenter would be faced with an extraordinary data storage and handling requirement.

Another aspect of the problem is one of spacecraft stability. The jitter requirement for the SSO is very tight (0.015 arc-sec/sec with a pointing accuracy of 0.5 arc-sec). The ability of the Sortie Lab to accomplish these taks is marginal at best; it is certain that several groups of ASF instruments would not be permitted to slew from one look direction to another during solar operations.

Table 5-3 illustrates the SSO instruments field-of-view requirements which are not consistent with the ASF requirement for full solar disc data. Table 5-4 provides a comparison of the spacecraft attitude requirements. As can be seen in Table 5-4, the ASF instrumentation would generally be in motion in order to compensate for spacecraft orbital motion, which would be detrimental to the solar physics data. The Orbiter pointing capability is expected to be sufficent for the ASF facility, but the SSO will require special image motion compensators as well as one more stage of pointing control beyond the experiment-pointing base.

Other areas of incompatibility are the desired orbit capability and the cleanliness requirements. The SSO facility cannot properly operate near 185km (100 nmi) because of atmospheric absorption. The SSO requires

Table 5-3. Solar Shuttle Observatory Instrument Field of View

Instrument	FOV	Purpose
Coronograph	15^{o}	1 sec resolution of solar prominences at the solar limbs. Flares at these locations are less important to earth phenomena than flares from more nearly central locations.
Flare Detector	Full disc	Identify and study active regions and flares.
EUV Spectrograph UV Spectroheliograph X-ray Spectroheliograph Visible Spectroheliograph, etc.	5 min 4 min 4 min 4 min	Scan over the disc and sum the results from ~64 FOV elements; i.e., 10^{6} words. These would provide spectral line profiles and total flux data.

Table 5-4. Comparison of Requirements

System	Pointing Accuracy	Jitter	Scan Rate	Total Instrument Weight
Basic	0.009 rad ± 0.5 deg	NA		
CMGs	0.003 rad 1.0 min	NA		
Standard Experiment Pointing Base (five)	0.005 mil rad ± 1 sec	NA		
Atmospheric Sciences Facility				
Main Instrument Group	0.0009 rad 3 min	0.01 mil rad/sec 2 sec/sec	.09 rad/sec 5 deg/sec	900 kg 2000 lb
Previewer	0.00175 rad 6 min	.0003 rad/sec 1 min/sec	.00058 rad/sec 2 min/sec	270 kg 600 lb
Solar Instrument	0.0053 rad 20 min	.0009 rad/sec 3 min/sec	0.02 mil rad/sec to 0.05 mil rad/sec	90 kg 200 lb
Shuttle Solar Observatory				
Pacing Instruments; e.g., Main Telescope	0.115 mil rad 24 sec	0.000073 mil rad/sec .015 sec/sec		5000 kg 11,000 lb
Crystal X-ray Spectroheliograph	0.005 mil rad .5 sec	0.001 mil rad/.2 sec .2 sec/.2 sec	0.0 mil rad/.2 sec 1 sec/.2 sec step raster	

500-km (270-nmi) orbit thus reducing performance margin for instrumentation which can be carried. The ASF can operate near optimum at 185 km (100 nmi). Additionally, the SSO requires the orbit to provide mimimum eclipse time; the ASF would utilize that orbit once but, for subsequent missions, would require different launch times. The SSO also has a stated requirement for contamination care that would seriously impact the cost of preparing the ASF instrumentation.

The most basic compatibility problem is one of data collection. The solar physicist wants high spatial resolution to follow the growth and decay of very small active regions on the sun. He would select those portions of the solar spectrum which provide information on solar processes. These specific spectral lines would be monitored in very high resolution in order to measure line widths which permit the calculation of temperatures and velocity fields. In contrast, the atmospheric scientist wants the total radiance from the entire solar disc that falls upon the earth. The spectral information required is much more coarse, since the atmospheric absorption of solar energy is due mostly to continuum absorption processes. Those relatively few cases of absorption resonance or accidental resonance for which solar line widths, shapes, and intensities are required are in general found at wavelengths of lesser interest to the solar physicist. Consequently, we find that the spatial resolution, time history data, and spectral data required by the two facilities are not complementary.

The two facilities might still occupy the same Sortie Lab as two strangers might occupy the same apartment building; they would share the same plumbing but would not reinforce one another's productivity. In fact, each one interferes with the other. The incompatibilities are summarized below:

1. The SSO separately records 64 FOV elements and $\sim 10^6$ resolution elements which require summation for ASF purposes.
2. The wavelengths chosen for observation by the solar physicist for solar studies may be irrelevant to ASF purposes.

3. Time sharing between the ASF "scan and sum" mode and the SSO "set and record each spatial element" mode is impractical on a seven-day mission since the main ASF experiments require simultaneous solar disc data.

4. The high resolution elements of SSO experiments are very costly and not required by ASF. SSO instrumentation is essentially wasted when used for ASF purposes rather than for collecting solar physics data.

5. The SSO experiments ought to all be flown as a group including the very large visible region telescope; the combined payload cannot all fit in a single Sortie and reach 500 km (270nmi).

6. The local time of ascending node acceptable to the SSO (maximum sunlit orbit) in a 55 degree inclined orbit is only one of several orbits the ASF would like to utilize in subsequent missions. If given a life expectancy of six missions, only one of these is compatible with the SSO.

C. THE MODIFIED ATMOSPHERIC SCIENCES FACILITY

In line with the philosophy expressed by the report of the Atmospheric and Space Sciences working group, (Ref. 20) the scope of the ASF was expanded to include active experimentation with the environment, more emphasis on solar-terrestrial relationships, and provision for monitoring stratospheric and mesospheric minor constituents.

The objectives chosen for the expanded Atmospheric Science Facility are as follows:

1. Established the effects of solar variability as a function of wavelength

 Photoionization rates
 Photodissociation rates
 Temperature and resultant chemical equilibrium changes; e.g., O_3 ion density

2. Establish remote sensing techniques including the addition of a vertical component to satellite data

 Composition and its variability
 Aerosols
 Pollutants

3. Investigate active chemistry with the upper atmosphere

 Determine the exited-state population of the ambient atmosphere
 Determine the role played by minor constituents in the chemical equilibrium

4. Determine diffusion rates

 Establish the eddy diffusion concept on a firm experimental basis
 Establish the facts about local contamination around a manned spacecraft
 Establish the size and dynamics of the spacecraft wake, bowshock, etc.

In order to accomplish these objectives, instrumentation has been added to the ASF as listed in Table 5-5. The most important departure is the slave satellite which permits simultaneous "in situ" measurements with its own instrumentation and absorption, resonant scatter, or emission measurements with the Sortie Lab instrumentation. This permits a real evaluation of the possible contaminant cloud surrounding the Sortie, the effect of the Sortie wake, etc. The slave Satellite, which carries both corner reflectors and laser detectors, can also be used for absorption measurements between itself and the Sortie Lab independent of the solar-zenith angle. The basic model of the slave adopted for this study is the Atmosphere Explorer C spacecraft (Figure 5-1). It should be noted that the plasma lab Sortie studies also contemplate use of a slave satellite.

The other instrumentation worth special note is the Light Detection and Ranging (LIDAR) unit which is a laser utilized for studies of aerosol scattering. With the LIDAR and the UV laser, it is likely that remote sensing of lower atmospheric pollutants and minor constituents can be studied.

It is not the purpose of this report to define the instrumentation in detail but rather to assess the requirements that a Sortie payload places on an integration facility and on orbital operations. Therefore, the modified Atmospheric Sciences Facility is not developed further.

Table 5-5. Added Instrumentation for the Modified ASF

	Instrument System	Purpose	Range and Sensitivity	Weight (kg)	Dimensions (m) L	W	H
1.	SLAVE Sub Satellite	Platform to study atmospheric chemistry, ion chemistry, diffusion rates, solar photoionization, composition, and density. (Design very like NASA atmosphere Explorer satellite C, D, & E; however, solar cells deleted for short mission and equipment list modified.)					
	a. Neutral Mass Spectrometer		1-60 AMU* $1/10^{-6}$	10	0.5	0.2	.2
	b. Ion Mass Spectrometer		1-60 AMU $1/10^{-6}$	9	0.5	0.2	.2
	c. Electro Langmuir Probe		0-50 eV 0.05 eV	4	0.2	0.1	.1
	d. Corner Reflector		UV-visible	4	1.0	1.0	.0
	e. UV Laser Detector		1000-1700 Å	10	0.1	0.1	.1
	f. Chemical Release Mechanism		NA	10	1.0	1.0	.0
	g. Large C_dA/M Factor		NA		NA	NA	NA
2.	Gas chemical release mechanism	Atmospheric chemistry studies.	Ba, Na, TMA, No	50	1	0.3	.3
3.	Laser Ranger	Determine distance, r and dr/dt to the SLAVE satellite.	Visible	15	1	0.2	0.2
4.	Tunable UV Laser	Absorption studies with the SLAVE.	1150-1800 Å	20	1	0.3	0.2
5.	Fabry-Perot Interferometer	Measure the temperature of emitting species, including effluents from the SLAVE.	Visible	35	1	1.5	0.6
6.	LIDAR	Measure aerosol scattering in the atmosphere below the satellite.	Visible and near 2500 Å	10	0.5	0.3	0.3
7.	Flare Detector	Signal to crew that a flare is occurring, locates flare position and records x-ray intensity.	3-25 Å	20	1	0.3	0.3
8.	Topside Sounder	Determine electron density and temperature remotely from the SORTIE module.	10^2 - $10^6 cm^3$	10	0.3	0.2	0.2

AMU: Atomic Mass Unit

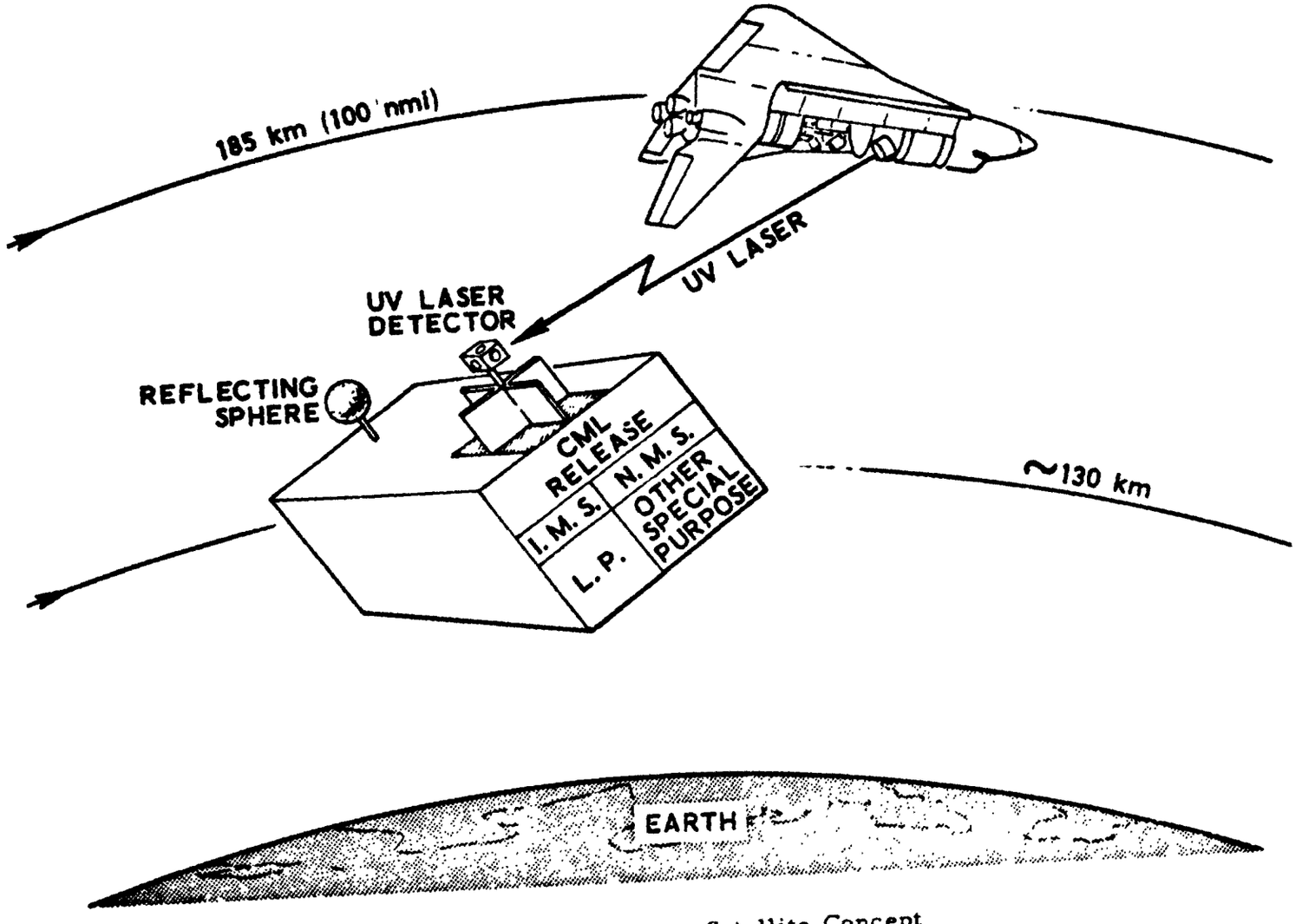


Figure 5-1. A Slave Satellite Concept

D. INTEGRATION AT A SORTIE CENTER

The concept of the Sortie payload system is that as a universal Sortie Lab module and pallet become available at an integration center (probably a launch site), a pre-built, tested, qualified, and calibrated group of instruments at the integration site is ready for mating to the pallet. The interface is a simple design with no complex alignment procedures required.

The packages and the operations required to integrate them onto the Sortie Lab pallet are listed in Table 5-6. The man-hour estimates are based upon actual operation times and do not include possible delays between operations due to scheduling of people, documentation, quality assurance checks, or on-the-job training. An increase of 50 percent in time is sometimes required, depending upon the contractor and operational philosophy of the management group. The operational times listed are considered realistic; they are based on current experience uncrating and checking existing satellite instrumentation. The time required for installation of consoles in the Sortie Lab is a reasonable estimate based on design expectations. It is possible that this operation could take much longer if the interface cabling design were not carefully worked out in advance. The times indicated should be conservative for a Sortie Lab integration after the process has become routine.

E. ORBITAL OPERATIONS

A series of typical experiments has been blocked out for the purpose of helping the systems designer understand what the scientist would like to do. The operator duties in Table 5-7 are described as if they were performed by a scientist and/or a highly trained technician. The total work time is estimated at 200 hours over the 6-day period. The time lines are shown in Figure 5-2, and they are tight for two men because the average work day is 16.5 hours. This tight scheduling is based on a number of observations:

Table 5-6. Integration Data at a Sortie Center (Probably a Launch Site)

Payload Part	Notes	Task	Total Man Hours	No. of Men	Labor Description	Special Equipment
SLAVE Satellite	Arrives intact and operational from the Payload Test Center.	Unpack single crate, visually inspect.	4	2	Payload Center tech.	Crane.
		Place on SORTIE-pallet retainer-ejection system.	4	2	SORTIE tech.	
		Connect umbilical to SORTIE lab.	1	1	SORTIE tech.	
		Checkout satellite to determine that safe transit occurred and no shipping damage.	16	2	Payload Center tech and payload project engineer.	Scopes, etc., standard. Checkout using SORTIE receiving equipment.
	Flight configuration permits mass spectrometer to be placed in ram while Slave still retained.	Put all Slave Satellite retainer and ejector systems into launch configuration.	3	1	SORTIE tech.	
General Purpose Spectroscopy Unit (GP)	Arrives in two crates. [Instrument barrel (see Martin) and gimbal mount.] Instruments have previously been calibrated, bore sighted, etc.	Unpack crates, visually inspect.	6	2	Payload Center tech.	
		Place gimbal system in airlock unit.	6	2	SORTIE tech.	Cranes, etc.
		Place instrument barrel into gimbal.	8	2	Payload tech.	Use SORTIE lab operational equipment for checkout.
		Install electrical interface to SORTIE lab.	2	1	SORTIE tech.	
		Checkout instruments and configure for launch.	16	2	Payload tech. and project engineer.	Not a calibration procedure.
Solar Monitor Unit (SM)	Arrives in two crates (instrument barrel and gimbal mount).	Unpack crates and visually inspect.	6	2	Payload Center tech.	
		Install gimbal system.	6	2	SORTIE tech.	
		Install instruments on gimbal.	8	2	Payload tech.	SORTIE lab operational equipment for checkout. Not a calibration procedure.
		Install electrical interface to SORTIE lab.	2	1	SORTIE tech.	
		Checkout instruments and configure for launch.	16	2	Payload tech. and project engineer	
Search or Preview Unit (Search)	Arrives intact and operational in two crates (gimbal mount and instrument barrel).	Unpack crates and visually inspect.	4	2	Payload tech.	
		Place gimbal system in SORTIE pallet.	6	2	SORTIE tech.	
		Place instrument barrel into gimbal.	8	2	Payload tech.	
		Install electrical interface.	2	1	SORTIE tech.	
		Checkout instrumentation and configure for launch.	12	2	Payload tech. and project engineer	
Other Chemical Release, Topside Sounder	Units are self-contained.	Unpack, bolt onto SORTIE pad, install electrical interface.	4 1	1 1	SORTIE tech. SORTIE tech.	
		Chemical release mechanism (spring ejector) armed and configured for launch.	1	1	SORTIE tech.	

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Table 5-6. Integration Data at a Sortie Center (Probably a Launch Site) (Continued)

Payload Part	Notes	Task	Total Man Hours	No. of Men	Labor Description	Special Equipment
SORTIE Lab Instrumentation	These units arrive as parallel units to the experiments, i.e., a console for each of the subgroups of experiments. These are mounted in standard SORTIE defined consoles that bolt into the SORTIE Lab directly.	Unpack and inspect all.	32	4	Payload tech.	
		SLAVE Satellite Console installed	8	2	SORTIE tech.	
		SLAVE Satellite Console electrical interface installed.	8	2	SORTIE tech.	Checkout console provided by Payload Center.
		SLAVE Satellite Console checkout.	8	2	Payload tech.	
		General purpose spectroscopy console installed.	8	2	SORTIE tech.	
		General purpose spectroscopy console electrical interface.	8	2	SORTIE tech.	
		General purpose spectroscopy console checkout.	8	2	Payload tech.	Checkout console provided by Payload Center.
	Checkout of these items involves placing of known signals at their inputs and monitoring their outputs. A special test console built at this Payload Center is the quickest, most efficient device for this.	Solar monitor console installed.	8	2	SORTIE tech.	
		Solar monitor console electrical interface.	8	2	SORTIE tech.	Checkout console provided by Payload Center.
		Solar monitor console checkout.	8	2	Payload tech.	
		Search unit console installed.	8	2	SORTIE tech.	
		Search unit console electrical interface.	8	2	SORTIE tech.	Checkout console provided by Payload Center.
		Search unit console checkout.	8	2	Payload tech.	
		Chemical release console installed.	2	1	SORTIE tech.	
		Chemical release console electrical interface.	1	1	SORTIE tech.	
		Chemical release console checkout.	1	1	Payload tech.	
		Data storage and retrieval unit. This may be a general purpose device of SORTIE specs to which all consules are designed to interface. Consequently, it may already be onboard. Checkout and electrical interface times above include the interface to the storage bank installation	6	2	SORTIE tech.	
	This schedule permits no repair, trouble shoot, etc. Slop for this must be additional					

Table 5-7. Orbital Operations

Operation No.	Operation Title	Description	Operator Duties	Rough Cut Estimates		Frequency of Operation	Units Used
				Setup Time	Time to Complete		
1	Post-Launch Checkout	After flight crew gives go ahead, turn on all instruments and check out systems. (1) SLAVE on, attitude sensors on, but thrusters disabled. (2) General purpose unit on, check monitor voltages and internal calibrate values. (3) Solar monitor on, check internal calibration values and monitor housekeeping voltages. (4) Search unit on, monitor internal housekeeping and calibration values.	Turnon, compares monitor values to a check list, evaluates problems, if any, and possible fixes. Decides to continue program or modify it if required. Checks with Ground for latest solar and magnetic storm predictions Evaluate first data.	N/A	4 hr	1 per mission	All
2	SLAVE-On Board Data	The SLAVE satellite is left on with attitude thruster system disabled. The mass spectrometers and Langmuir probes take data while slave still retained on SORTIE. This provides outgassing data on board the large SORTIE. This continues 6 hours before the slave is ejected from SORTIE.	Evaluate data. If the background pressure is too high, he can decide not to operate high voltages or can protect optics of other mission equipment. Operator not occupied full time. Part of Op 1 can be simultaneous.	15 min	6 hr	1 per mission	SLAVE, SM
3	Wake Investigation	Release SLAVE at 0.5 ft/sec. Umbilical or command can enable SLAVE thrusters. (1) As drifts away, measure extent of SORTIE outgas cloud. (2) Using very low level thrusters ($\Delta V \sim 0.25$ ft/sec granularity), maneuver SLAVE across SORTIE wake at distances of 5, 20, and 100 m behind SORTIE, repeat in day and night orbit portions. (3) Permit to drift below and ahead of SORTIE. Used with UV laser, laser ranger, solar monitor, and general purpose unit to record UV laser returns. These are due to particulates or condensed matter "outgassing."	One crew member will be required ~full time to "drive" the slave. The other will monitor data, operate the laser ranger on the search unit and other equipment. Solar unit is also monitoring total UV flux. Flare alarm unit on.	30 min	9 hr	3 (early, middle, and late mission to measure gas cloud rate of dispersal)	SLAVE, Search, GP, SM
4	Absorption (Compare in situ to optical values of composition)	Point the Search unit of SLAVE. SLAVE kept within a km or so of SORTIE altitude in first sequence. UV laser tuned and pointed to SLAVE (has a UV detector). Laser fine tuned for max absorption spectra and column density of atmospheric species. Change in absorption with range permits effect of optical degradation to be evaluated. Depending on range over which laser can be tuned, set for absorption in N_2, O_2, and O_2, N_2. This operation is performed at 10° increments of latitude, day and night. The operation is repeated six times during mission and more often if k_p or flare changes occur. The SLAVE mass spectrometer data are compared for consonance with path absorption data to evaluate in situ performance against optical performance. Four of the six operations should be with the SLAVE at different altitude levels. If lasers can put out millijoules, then can operate over several 10's of km. The relative attitude of the sun vs slave UV detector field of view is critical.	Solar unit automatically tracks sun. Search unit can automatically track SLAVE with cornor cube-laser ranging unit as a sensor. General purpose unit can be slaved to search unit (GP unit has a fairly large field of view) or manually tracked. Operator tunes laser for absorption feature desired. Second operator drives GP unit and, as necessary, controls SLAVE position.	1 hr	4 hr each of 6 operations	½ per mission	Search, SLAVE, GP, SM

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Table 5-7. Orbital Operations (Continued)

Operation No.	Operation Title	Description	Operator Duties	Rough Cut Estimates: Setup Time	Rough Cut Estimates: Time to Complete	Frequency of Operation	Units Used
5	Calibration of Remote Sensors	Operation No. 4 is a part of this one, i.e., the operation at several altitudes. Add the monitoring of resonantly scattered laser beam and detection by the GP unit. The backscattered intensity is proportional to the number density. The data derived from this is also compared to the SLAVE mass spectrometer. The emission from N_2 excited by sunlight and ambient photoelectrons has been suggested as a monitor of N_2 density, therefore, record emission at 3371 and 1041 Å with GP unit in conjunction with N_2 mass spectroscopy as a function of SLAVE altitude.	See above. Also operate the topside Sounder for electron density data. This operation is designed to compare in situ with remote techniques and is not a source of a remote sensing data base. This can be worked in with other operations to avoid XC maneuvering of the SLAVE.	Setup for 4 is okay	4 hr additional to above	2 per mission required. If time, should repeat during a magnetic storm.	GP, SM, SLAVE, Sounder
6	Earth Limb Profiles	Limb profiles in all wave lengths are obtained with GP unit. The purpose is to map earth limb as f(h) and to note changes that may occur with flares, magnetic activity, latitude, or local time. Used to check models of atmospheric emissions.	The GP unit is scanned in both wavelength and altitude.	15 min	4 orbits, 6 hr	1 min, again if flare	GP, SM
7	Heat Balance	Survey Nadir emissions in all wavelengths with GP unit. SM unit to record solar spectra, and selected line profiles can be set to be ~ automatic.		30 min	4 orbits	1 per mission, may repeat if flare occurs.	SM, GP
8	Aerosol Monitor	Using LIDAR scattering and solar scattering measure the reflected light from aerosols and from Mie scattering. Using six or more selected wavelengths determine the wavelength dependence and from the LIDAR a measure of height of layer. These will be interpreted along with the UV Nadir data of Operation 7. LIDAR part of experiment performed on night side of orbit too. These measurements will also be taken as a function of sun angle.	Control search unit and GP unit to view some portion of space. Set angle relative to sun. Interpret data for surprises and provide alternate procedures as desired.	30 min	16 orbits to get geographic and time distribution.	1	SM, GP, Search
9	Atmospheric, Temperature, and Velocity Pattern	Using Fabret-Perot instrument - survey the the thermal breadth and Doppler shift of atmospheric line emissions (e.g., 5577 and 6300 Å). Obtain day and night data in quite and disturbed times.	Pointing of the GP unit to provide the orbital Doppler shift bias desired and to select filters required. Monitor data for surprises and provide alternate procedures as required.	30 min	2 orbits	2 per mission	GP, SM
10	UV Scatter	If Operation 5 shows laser strong enough and the particulates from SORTIE are few enough shine UV laser in nadir direction, scan GP unit along beam (units are placed at opposite ends of the pallet) and monitor scattered intensity. Determine total density from Mie or Rayleigh scattering and composition from resonant wavelengths. This is also a monitor of SORTIE outgassing of particulates. Do this as a data collection exercise (as distinct from 5).	Evaluate data from Op 5, control GP pointing.	30 min	2 orbits	1 per mission	GP, Search, SM

Table 5-7. Orbital Operations (Continued)

Operation No.	Operation Title	Description	Operator Duties	Rough Cut Estimates		Frequency of Operation	Units Used
				Setup Time	Time to Complete		
11	Chemical Release	Eject chemical release container. When ~1 km away it will release the chemical (several projectiles containing NO, Ba, TMA, etc., release one at a time). Using the GP unit and LIDAR unit, monitor the expansion, the brightness, and the relative emission of one wavelength and another. Monitor electron density with ionosonde. As one of last operations before retrieving SLAVE, fly slave through a NO cloud to monitor ions and chemical species within the cloud.	Point GP unit, select wavelength to monitor, evaluate data for surprises. May modify planned sequence for ejecting chemicals.	30 min	30 min, each in sequence over ~3 orbits	10, probably	SM, GP, SLAVE, Sounder and release
12	Solar Intensity	Obtain a spectral scan of the sun in all wavelengths.	Switch SM unit to scan mode, set scan speed	10 min	40 min	3 per day	SM
	Line Profile	Obtain profiles of selected solar emission lines.	Switch SM unit to scan mode, se scan speed.	10 min	40 min	3 per day	SM
13	Flare Data	When flare alarm sounds go to Operation 12, followed by 14. Repeat 12 every 2 hr for 12 hr. Also, put GP unit into Operation 7.		10 min	40 min	every 2 hr	SM, GP
14	General Search	General preview monitor obliquely down and forward of the SORTIE. The sounder and filter photometers to collect a data base of usual and warning of unusual events ahead of the spacecraft. This mode will be used any time the search unit is not being used in special Operations 1 through 11.	Routine measurements require only occasional monitoring to assure proper functioning.				
15	General GP	This mode will be nadir scans as a function of wavelength. The line profiles of O, H, and N_2 emissions will also be collected at all times not utilized for special Operations 1 through 11.	See Operation 14.				
16	General SM	Broad band UV intensities will be monitored at all times the SM unit is not involved in special Operations 1 through 13. The operator will pick one of a few lines for monitoring with the high resolution line profile spectrometer. Different wavelengths may be chosen for monitoring based on data from Operation 12 data.	See Operation 14.				
17	Return Prep	Prepare to return, strap down all gear, lock gimbals, turn off equipment, etc.					

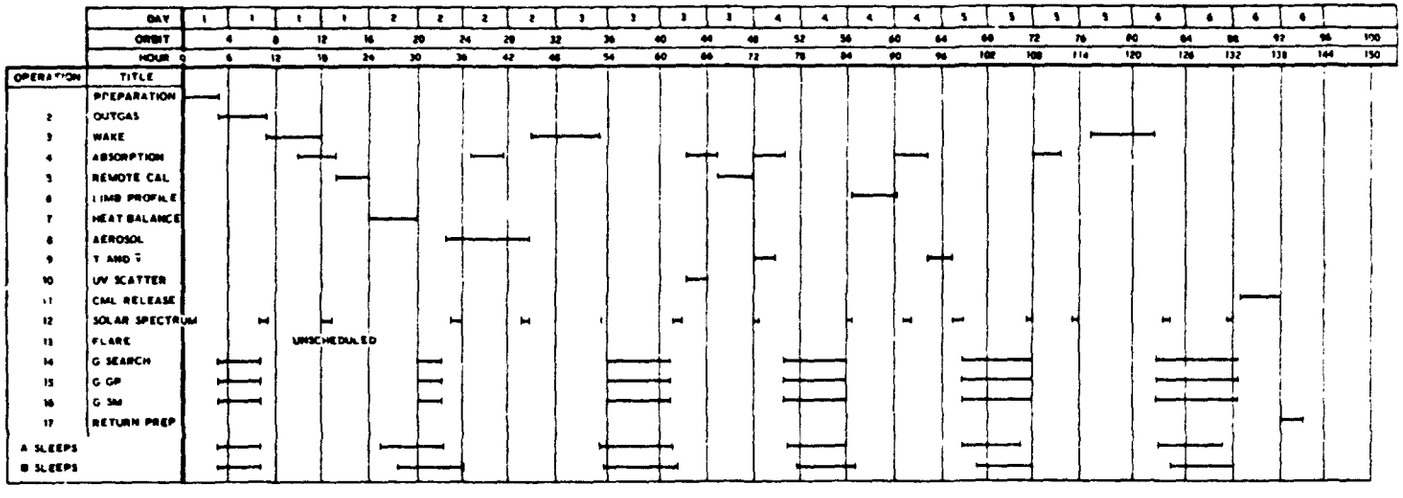


Figure 5-2. Time Lines Measured from Experiment Start (Not Launch Time)

1. Scientists spend very long hours in the laboratory when exciting experiments are reaching fruition. It is not likely they will change character and work an eight-hour day on a seven-day mission, and it is equally unlikely they could be taken [illegible] from the consoles during their waking hours.

2. This mission profile is presumed to be a first or survey ASF mission in which many techniques will be tested for future development where legitimate and useful data for scientific purposes are also obtained.

3. Each of the instruments is sufficiently automated that the scientist must spend most of his time in monitoring data, in quick analysis, and in selecting detailed modes of operation. He should not be required to manually operate equipment normally. Therefore, time for eating, coffee breaks, etc. is available within the allotted time elements, and experience shows that many meals are taken at the desk or in the laboratory during intense data-recording runs.

The addition of a third man would reduce the average work day to 11 hours. He would also be useful to relieve some of the strain from the other two and provide a third party to discuss data which can be very beneficial. The time lines shown should be considered the maximum schedule possible.

The operations listed are not necessarily those for an actual flight, but they should be useful to the system analyst in understanding the potential desires of orbiting scientists who may be expected to approach problems somewhat differently than astronauts.

F. GROUND OPERATIONS

The upper level integration flow cycle for the Shuttle, Sortie Lab, and experiment is shown in Figure 5-3. The ground operations for the Sortie experiment start at the lead center and end after landing and removal of the equipment. The operations where the experiment will interface with the Sortie Lab occur during experiment installation, orbital operations, and experiment removal. The experiment interfaces with the Sortie Lab, and the Sortie Lab interfaces with the Orbiter. The orbital configuration of the modified ASF is shown in Figure 5-4, and the experiment equipment list is shown in Table 5-9.

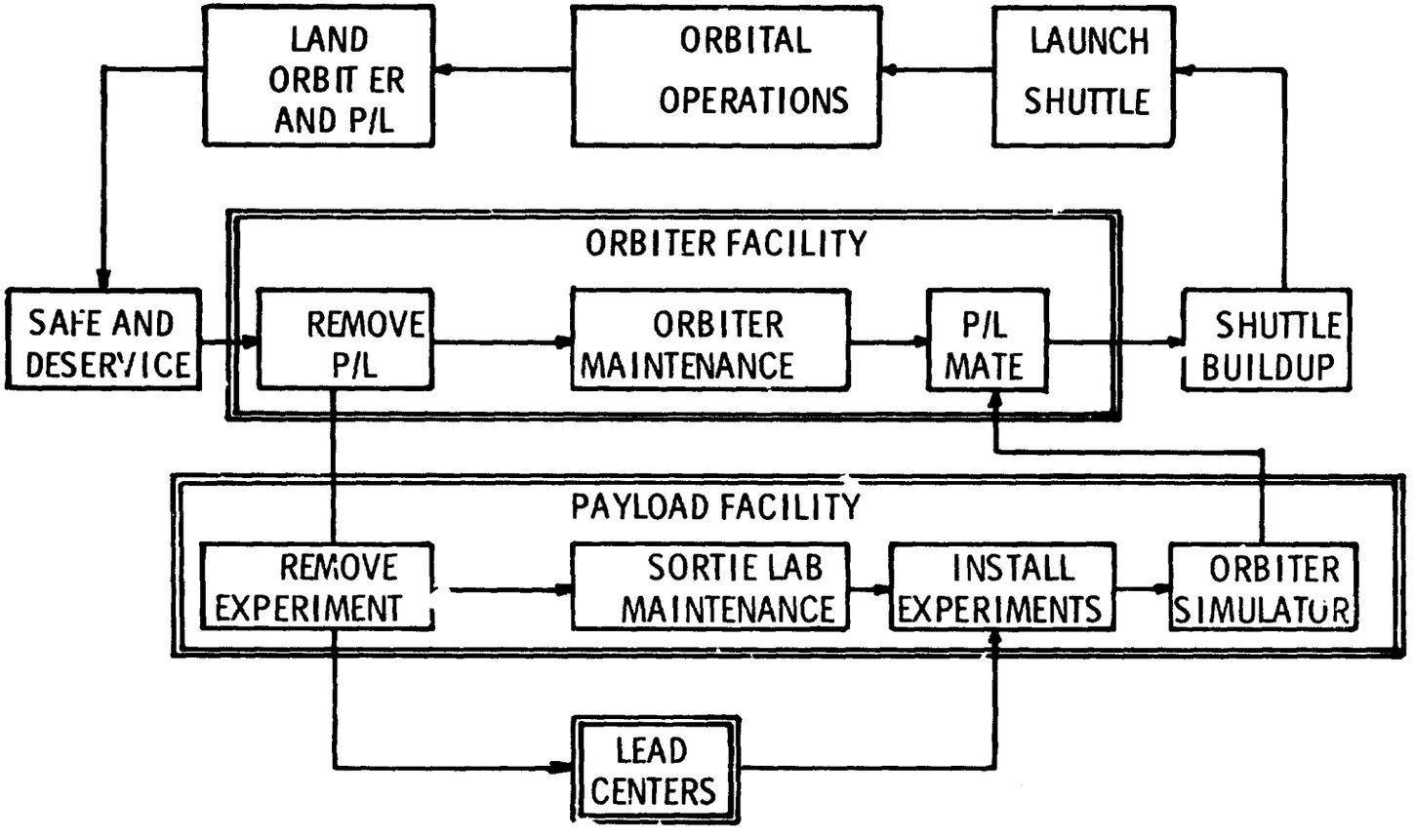


Figure 5-3. Pay. d Operations Flow Cycle

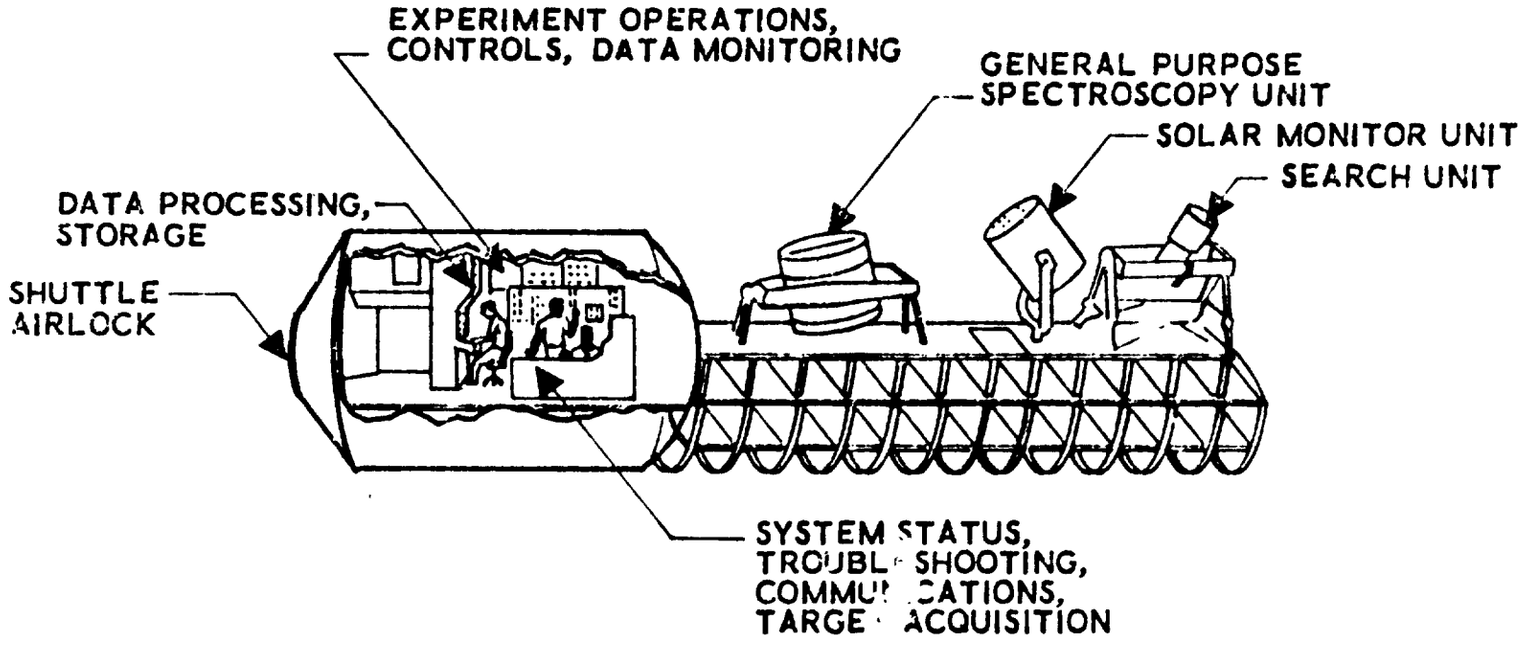


Figure 5-4. Modified ASF Orbital Configuration

Table 5-8. Experiment Equipment Main Assemblies and Complementary Equipment

- Gn'l Purpose Spectroscopy Unit
 - Gimballed Mount
 - Star Tracker
 - XUV Normal Incidence Grille Spectrometer
 - UV Visible/Near IR Spectrometer
 - High Resolution IR Interferometer Spectrometer
 - LN_2 Cooled Long Wavelength IR Interferometer Spectrometer
 - Filter Photometers
 - TV Display Camera
 - Laser Ranger
 - Tunable UV Laser
 - Fabry/Perot Interferometer
 - UV Documentation Camera
- Solar Monitor Unit
 - Gimballed Mount
 - Solar Tracker
 - XUV-UV Solar Intensity Monitor
 - Grazing Incidence Spectrograph
 - Normal Incidence Spectrograph
 - Echelle Spectrograph
 - Flare Detector
- Search Unit
 - Gimballed Mount
 - Filter Photometer
 - Zoom TV Camera
 - Light Detection and Ranging
- Slave Satellite
 - Ejection System
 - Neutral Mass Spectrometer
 - Ion Mass Spectrometer
 - Electron Langmeir Proble
 - Corner Reflector
 - UV Laser Detector
 - Chemical Release Mechanism
- Gas Chemical Release Mechanism
- Control and Display Electronics
- Wire Harness Assemblies
- Bulkhead Fittings
- Expendables
 - Polaroid Film
 - Strip Chart Recorder Paper
 - UV Documentation Camera Film
 - LN_2 Bottle and Lines

The lead center serves as the starting and focal point for all of the experiment equipment. Individual instruments and their controls will arrive from manufacturers, research agencies, or other sources. The lead center can mount the instruments in the appropriate gimballed main assembly and their controls in the control consoles. The control consoles will then be attached to the Sortie Lab Module standard fittings. All experiment equipment can be physically mocked up in the flight configuration. The Sortie Lab data collection and storage system, the electrical power, and environmental control systems should be available for integration checkout. Interconnecting electrical harnesses should be fabricated and certified. Preliminary mass properties data can be determined if required. The experiment equipment should be aligned, the calibration established, and a complete performance verification conducted with the Sortie Lab. It was assumed that the experiment is capable of performing on-orbit calibration so that ground calibration at the launch site would not be required.

Upon completion of the lead center operation, the experiment equipment would be disassembled to the main assembly level as listed in Table 5-8 for shipment to the launch site.

For integration of the experiment equipment with the Sortie Lab at the launch site, the installation of the instrument assemblies on the pallet and the control consoles in the Sortie Lab module should be done simultaneously on a no-interference basis. However, work in the Sortie Lab module should be planned serially as should the pallet installation activities. Using this approach, the operational sequence and time lines for the experiment installation and checkout operations are shown in Figure 5-5 for the Sortie Lab module and in Figure 5-6 for the pallet. It can be seen that about nine single-shift working days are required. The checkout times shown include verification of the interface to the data collection and storage equipment of the Sortie Lab module.

The checkout of support equipment items is defined as end-to-end type verification where known signals are inputted and the outputs are

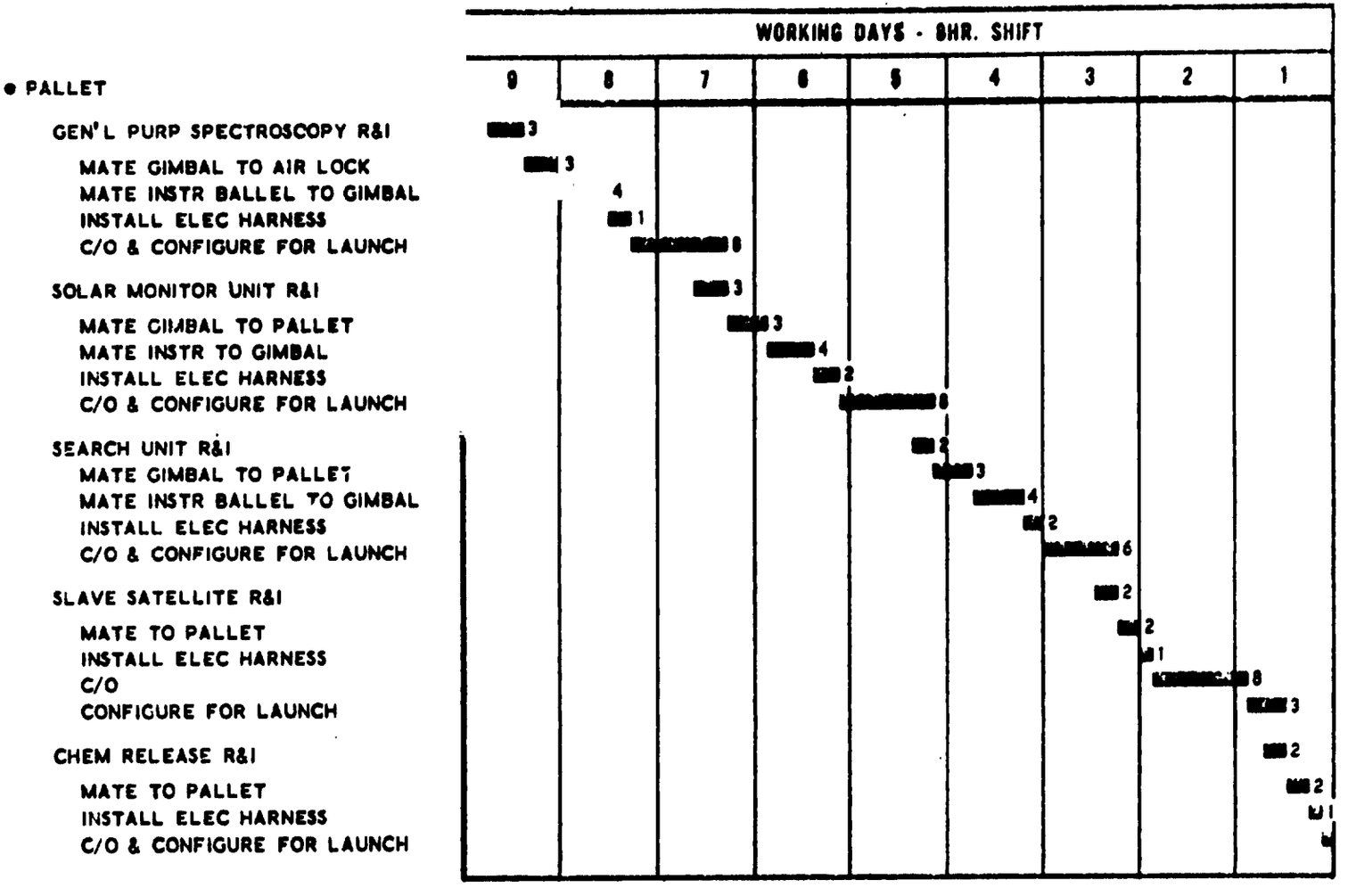


Figure 5-5. Instrument Integration Time Lines

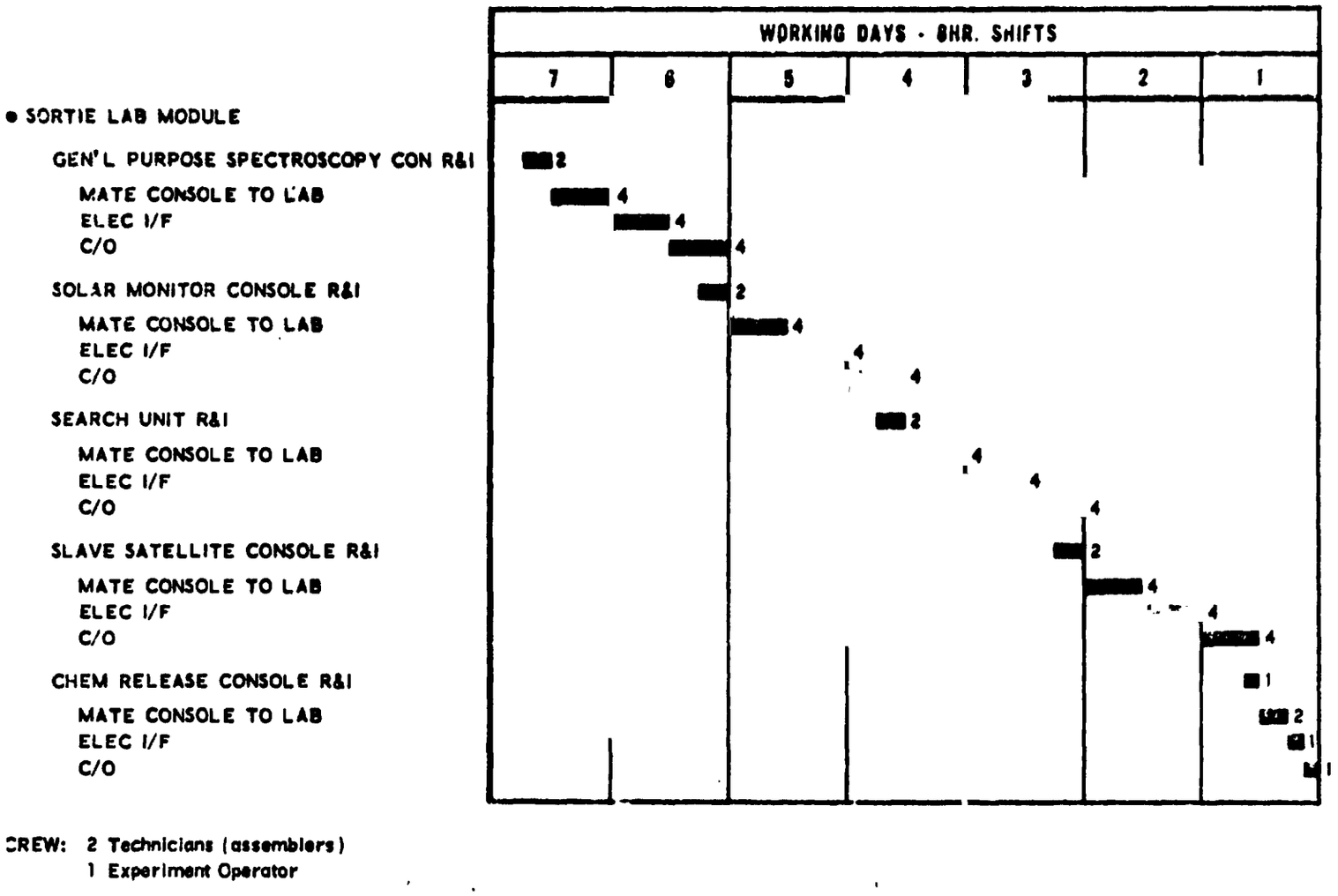


Figure 5-6. Control Consoles Integration Time Lines

monitored. A special test console (built and checked at the lead center) should be the most efficient approach. Special handling equipment to position the assemblies will be required. No unusual facility requirements are foreseen. Facility floor space needs for integration would be met by the 557m^2 (6000 ft^2) that have been identified in Reference 21. An overhead crane of 907-kg (2-ton) capacity with a clear hook height of 9.1m (30 ft) should be adequate.

The expendables and servicing items are conceived to be:

1. 40 rolls of Polaroid film
2. 10 rolls of strip-chart recorder paper
3. 1 roll of film for each UV documentation camera on the general purpose spectroscopy unit
4. One LN_2 bottle

The first three items can be stored during the integration activity. The LN_2 bottle should be installed just prior to installing the Sortie Lab to the Orbiter.

The personnel for this operation are two crews of two technicians each: one crew for control console installation in the Sortie Lab module and the second for instrument installation on the pallet. It is expected that four men would be augmented by the mission and payload specialists who would assist in checkout activities and problem resolution. Thus, a total crew of seven, intimately familiar with the experiment equipment and assisted by launch-base personnel, is anticipated. No further ground operations devoted to verifying experiments which are planned prior to launch.

After return from orbit, the Sortie Lab is removed from the Orbiter and transferred to the payload facility for removal of the experiment equipment. The facility and personnel requirements are the same for experiment equipment removal as for integration. Operations for removal are expected to be similar in that work will be done concurrently on the pallet and in the Sortie Lab module. The removal time is expected to be about the same as the mechanical installation time (with no checkout). On this basis, the time required for removal of the experiment equipment is five working days, sequenced and timed as shown in Figure 5-7.

- SORTIE LAB MODULE
 - REMOVE FILM, TAPE, CHARTS
 - REMOVE ELEC I/F
 - DEMATE CHEM RELEASE CONSOLE
 - REMOVE ELEC I/F
 - DEMATE SLAVE SATELLITE CONSOLE
 - REMOVE ELEC I/F
 - DEMATE SEARCH UNIT CONSOLE
 - REMOVE ELEC I/F
 - DEMATE SOLAR MONITOR CONSOLE
 - REMOVE ELEC I/F
 - DEMATE GEN'L PURPOSE SPECTROSCOPY CONSOLE
- PALLET
 - DEMATE GIMBAL
 - DEMATE GEN'L PURPOSE SPECTROSCOPY INSTR. BARREL
 - REMOVE ELEC I/F
 - DEMATE GIMBAL
 - DEMATE SOLAR MONITOR UNIT INSTR BARREL
 - REMOVE ELEC I/F
 - DEMATE GIMBAL
 - DEMATE SEARCH UNIT INSTR BARREL
 - REMOVE ELEC I/F
 - DEMATE SLAVE SATELLITE
 - REMOVE ELEC I/F
 - DEMATE CHEMICAL RELEASE
 - REMOVE ELEC I/F

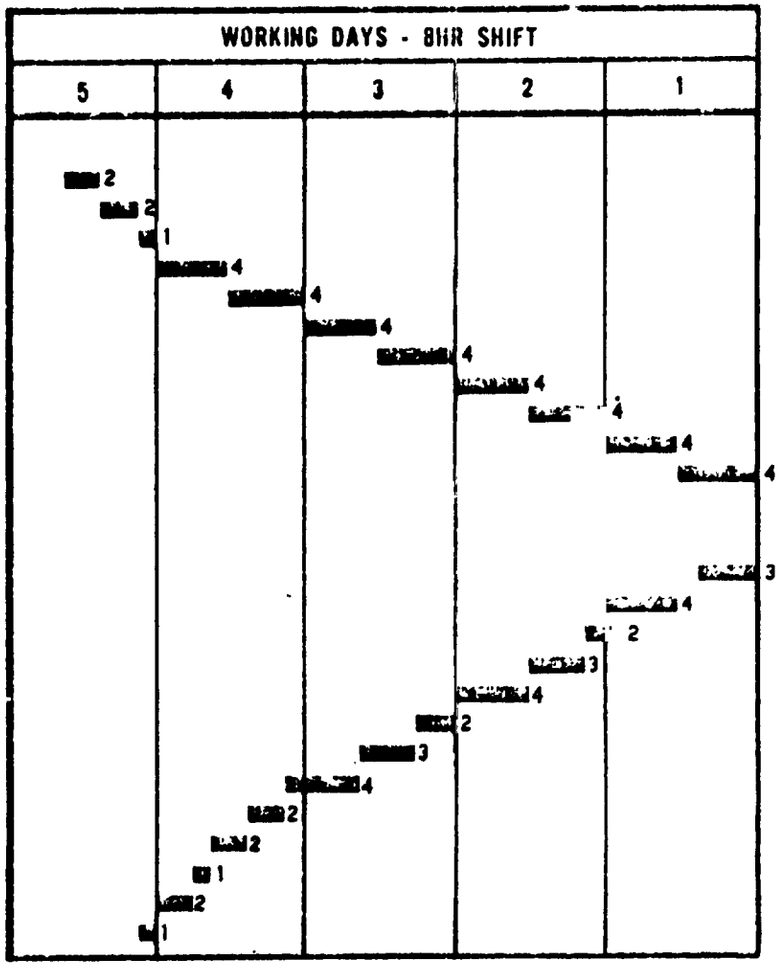


Figure 5-7. Experiment Equipment Removal

G. SORTIE PAYLOAD CAPABILITY

The combination of the Solar Shuttle Observatory and the Atmospheric Science Facility was determined to be incompatible because the two Sortie missions are not mutually supporting, have different orbital requirements, and interfere with one another's operation. Since these payloads were incompatible, it was necessary to develop a model Sortie payload for the study. For this model, the Atmospheric Science Facility was modified to incorporate more recent research requirements. The description of the modified version was provided in the scientific orbital operations for use in modeling a standard mission.

The modified Atmospheric Science Facility representing the standard mission was found to be operationally compatible with the existing definition of the Sortie Lab (Ref. 22) and Space Shuttle (Ref. 23). An operational comparison of the Shuttle and Sortie Lab capabilities and the experiment requirements are summarized in Table 5-9. The resulting compatibility appears due to the nature of the equipment design concept. To provide this compatibility, the experiment equipment should include the following design features:

1. Experiment hardware interfaces with the Sortie Lab only
2. On-orbit alignment and calibration capability
3. Experiments delivered assembled at the launch site in flight-ready condition
4. Minimum servicing and expendables

Table 5- 9. Evaluation of Operational Compatibility

Item	Shuttle Sortie Lab	Experiment Requirement
Shuttle (Level I)		
Number of Mission Specialists	2 to 4	2 to 3
Mission Duration	7 days	7 days
Payload Definition	Self sufficient systems	No Orbiter Interfaces
Checkout	Payload test and checkout required prior to installation in Orbiter	All experiment equipment testing done prior to Payload installation in Orbiter
Turnaround	14 calendar days	No impact on any turnaround operation
Sortie Lab		
Mission Duration	6.5 days for on-orbit operations	6.0 days of on-orbit experiment operations
Crew Schedule	Simultaneous or staggered work/rest cycles of 12 hours on/ 12 hours off	Mainly simultaneous work/ rest cycles of 11 hours on/ 13 hours off for 3 crew
Ground Operations Time	Not specified	14 working days

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APPENDIX A. STRUCTURAL DESIGN

This appendix is an attempt to generate general design criteria and guidelines for low-cost flightworthy structure in terms of design safety factors by a rational approach which maintains adequate structural integrity. It is primarily a discussion on promising alternatives to procedures which are now almost traditional for weight-optimized structure development. In this form, it is intended to stimulate further cost reduction thinking when applied to specific designs to be developed by particular engineering and fabrication facilities.

The diversity of current viewpoints on the actual utility of safety factors and the proper numerical values which are appropriate for any particular mission will prevent unanimous agreement on any recommended values. The recommended safety factors resulting from this analysis are therefore recognized as being somewhat arbitrary, reflecting the judgment and experience of The Aerospace Corporation. They are intended as starting point values to be modified as low-cost experience is gained or as other data becomes available.

Results of this effort are also summarized in Appendix B in a form familiar to potential hardware developers. This appendix is typical of material supplied for inclusion in the work statement for a number of DOD programs and is in the form of sample general structural design criteria. The criteria are intended to form the basis of and provide direction for the detailed criteria to be developed by the structure subsystem developer.

A. ASSUMPTIONS

It is assumed that the requirements for safety and reliability of the payload structural subsystem do not decrease for Shuttle-launched payloads. Considerations of Shuttle and personnel safety alone require that the payload be designed to prevent damage or injury. Payload structural reliability (the characteristic which is defined here as performing an intended function)

also must be maintained throughout the specified design life due to the impracticability of performing on-orbit spaceframe repairs. It is assumed that it is not cost effective to purposely design structure to a specific failure rate because of the Shuttle trip costs. These assumptions are reflected in design goals of no performance degradation, rupture, or collapse during design life at limit load and the requirement to demonstrate the existence of specified design safety factors by test or appropriate analysis.

It is a basic premise that safety factors do not attempt to allow for overloading or any other anomalous environmental condition. If performance margins are necessary or desirable, appropriate adjustments should be made to the specified limit or design conditions.

The following development also leads to the implicit assumption that the purpose of design safety factors is to account for differences in strength capability between test and flight hardware. In the case of structure which is not test-verified, safety factors must be increased to encompass unavoidable uncertainly in analytical strength prediction.

B. ANALYSIS

1. GENERAL

To explore the cost reduction potential associated with over-designed structure, it is illustrative to first examine the function of safety factors as currently applied by the aircraft and aerospace industries. These industries share the requirements placed on structure of extremely high reliability and safety, because of the very high costs associated with failure, and minimum practical weight, because of relatively high transportation costs. These requirements have been met quite successfully through development and inspection programs which involve extensive analysis effort, complex fabrication methods, rigorous quality specifications, and static test qualification. Expensive, but cost effective, structure is the result.

If transportation costs become less sensitive to weight, as with Shuttle-launched payloads, it is possible that a heavier structure may be designed which is lower in cost. There are two basic ways that this type

of cost reduction may be accomplished: by under-optimization and/or by over-design. It is important to recognize that these are almost totally independent approaches to cost reduction. This study primarily addresses over-designed structure.

2. UNDER-OPTIMIZATION

Relaxation of weight constraints immediately suggests that the design and fabrication effort associated with weight optimization be reduced or even eliminated. Most of these cost-avoidance areas are obvious and have been adequately reported in the literature. Typical of these approaches is the use of less efficient structural materials (lower cost, higher weight per unit of load-carrying capability) and elimination of material removal in lower stressed regions of each structural element (less machining, chem-milling, lightening holes, etc.). Additional cost savings are realized through associated reductions in analysis complexity, development testing of exotic materials, problems with flaw sensitivity of highly tempered metals, and so on. All of the cost savings in the category of under-optimization are direct trades of cost and weight and do not generally improve safety or reliability.

3. OVER-DESIGN

Over-designed structure involves a general increase in strength by increasing the size of load-carrying structural elements. One rational way of allocating extra material through the structure is by the application of increased factors of safety. This approach will increase hardware cost and weight but offers potential overall cost savings by reducing development costs associated with qualification analysis and test. For example, if safety factors are made sufficiently high, it may be possible to eliminate static testing completely without sacrificing reliability and safety. Intermediate safety factors may allow simplified testing. The potential savings associated with over-designed structure can only be determined by detailed cost analysis of a particular design; however, a feeling for this potential may be obtained from a simple consideration of approximate costs.

Structural subsystem weight for lightweight payloads approximately 15 - 20 percent of the total payload weight. Even for very large satellite in the 5,000-kg (11,000-lb) category, a yield factor of safety of three could be achieved for less than a (0.2)(5,000)(3-1) = 2,000-kg (4,400-lb) increase. At a material cost of less than $42.00/kg ($0.96/lb) (aluminum alloys), a $4,000 investment offers the possibility of eliminating a static test and test hardware costing $100,000 - $200,000. Against this must be traded whatever additional effort is required, perhaps engineering analysis.

It is apparent that to continue this type of estimation it is necessary to determine factors of safety which will allow this or other approaches to structure qualification. It is also necessary to define more accurately the requirements placed on analysis and quality control by this approach, particularly if they differ from current requirements for test-qualified lightweight structure. To this end, an understanding of the real function of safety factors, as now utilized, is necessary.

4. SAFETY FACTOR UTILITY

Examination of the intended function of safety factors necessary for an understanding of their relationship to testing, quality control, and analysis is difficult because of the many interpretations currently popular within the industry. There is seldom agreement even within a single organization as to the function of safety factors. Various designers and analysts consider their purpose is to cover ignorance and/or error in an assortment of areas including hazard, load uncertainty, material uncertainty, process variations, and analysis inaccuracies. Insight into the actual role of safety factors may still be gained, however, by examining one of the several rational design approaches now being used. An ideal approach, from a weight, schedule, and reliability standpoint involves the following basic tasks:

a. Establish upper-bound loads including dynamic loading effects (limit loading)

b. Determine upper-bound critical stresses for limit loads and ultimate loads (product of limit load and safety factor). (This may be accomplished through a combination of rigorous analysis, conservative assumptions, and experimental analysis.)

c. Size each structural element using lower-bound element capability so that yielding does not occur below limit load and rupture does not occur below ultimate load. (Performance of this task presumes a knowledge of minimum material and assembly quality based on purchase specifications, process control, and inspection.)

d. Qualify the design by applying limit and ultimate loads to a prototype structure

e. Control the quality of all subsequent units to a level consistent with the specified design safety factors.

Figure A-1 is a simple illustration of the variables involved in a deterministic establishment of safety and reliability. The ordinates plotted are applied loads and load-carrying capability which have been normalized to limit loads. As such, they are safety factors by definition. Yield and ultimate factors of safety are shown in four general categories: those specified as design requirements, those theoretically attainable, those actually present in fabricated structure, and those which are demonstrated by static test. The range in theoretical safety factors is the result of expected variations in strength of the actual design where the upper limits correspond to "perfect" fabrication from materials which happen to have upper-bound strength characteristics. The lower limits correspond to the minimum strength which could result within the adopted material specifications, fabrications, fabrication methods, process controls, and inspection requirements. Theoretical safety factors are a characteristic of the design and its quality specification only and are not directly affected by any strength analysis which is conducted. If the design is based on analysis, however, an inadequate design may result from analytical errors. Actual safety factors are a characteristic of specific hardware and the conformance of quality control to quality specification. If the previous design tasks are conducted rigorously, as would be necessary for weight efficiency, several fundamental relationships may be deduced from the figure.

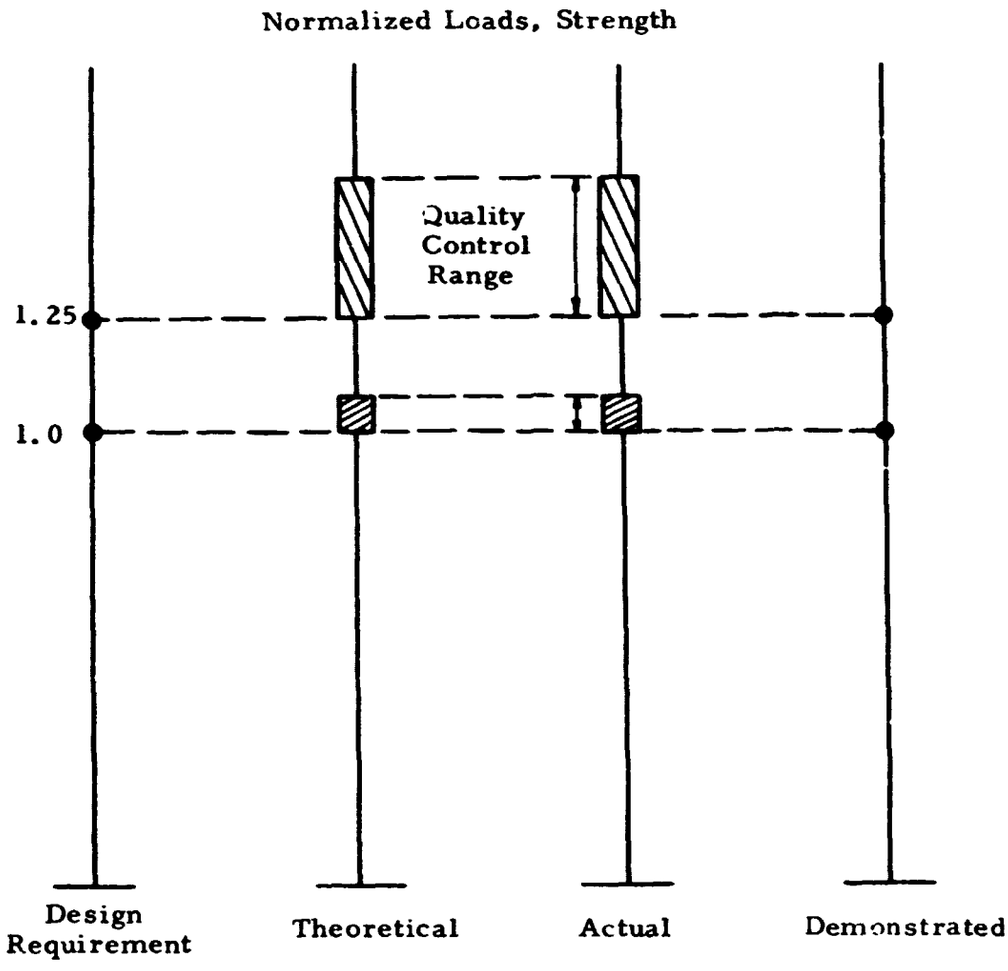


Figure A-1. Ideal Lightweight, Test-Qualified Design/ Safety Factor Relationships

A successful limit and ultimate load test of the prototype establishes the load-carrying capability possible with that particular design. The test article is probably damaged by the test, however, and its subsequent utility as a flight article is not known. Payloads designed to these safety factors normally require a dedicated test article to verify the design.

Structural capabilities of each untested flight structure will be generally different, resulting in the ranges indicated. The lower bound of these ranges is known only if there is adequate quality control and conservative stress analysis. This family of structures meets the established requirements; some exceed these requirements because of the application of conservative stress analysis and the unavoidable variations in fabricated hardware. In this ideal case, the real utility of a design safety factor is simply to provide an arbitrary, pre-specified element of safety in the event of overloading, without necessarily contributing to reliability. It does not compensate for uncertainty in limit load determination, quality control, or stress analysis. The mean safety and reliability for the family is higher than required, but by an amount which is not known. This is certainly a desirable feature, and if the family is large enough, it may be possible to statistically evaluate the consequences of errors or anomalies in determination of limit loads. Nevertheless, this additional safety is the result of material and process variations and design conservatism rather than the application of safety factors.

Turning away now from idealistic situations, it is possible to arrive at the relationship shown in Figure A-2 through a combination of the selection of an upper-bound structure for test, inadequate quality control, and/or stress analysis errors. If the test article happened to be the best of the family (quite possible with prototype hardware), the design requirements are not met except for the test article, and this fact will not be known prior to flight. This situation arises either because the upper-bound stress was under-determined or because the lower-bound capability was over-stated, but the end result is the same. Performance

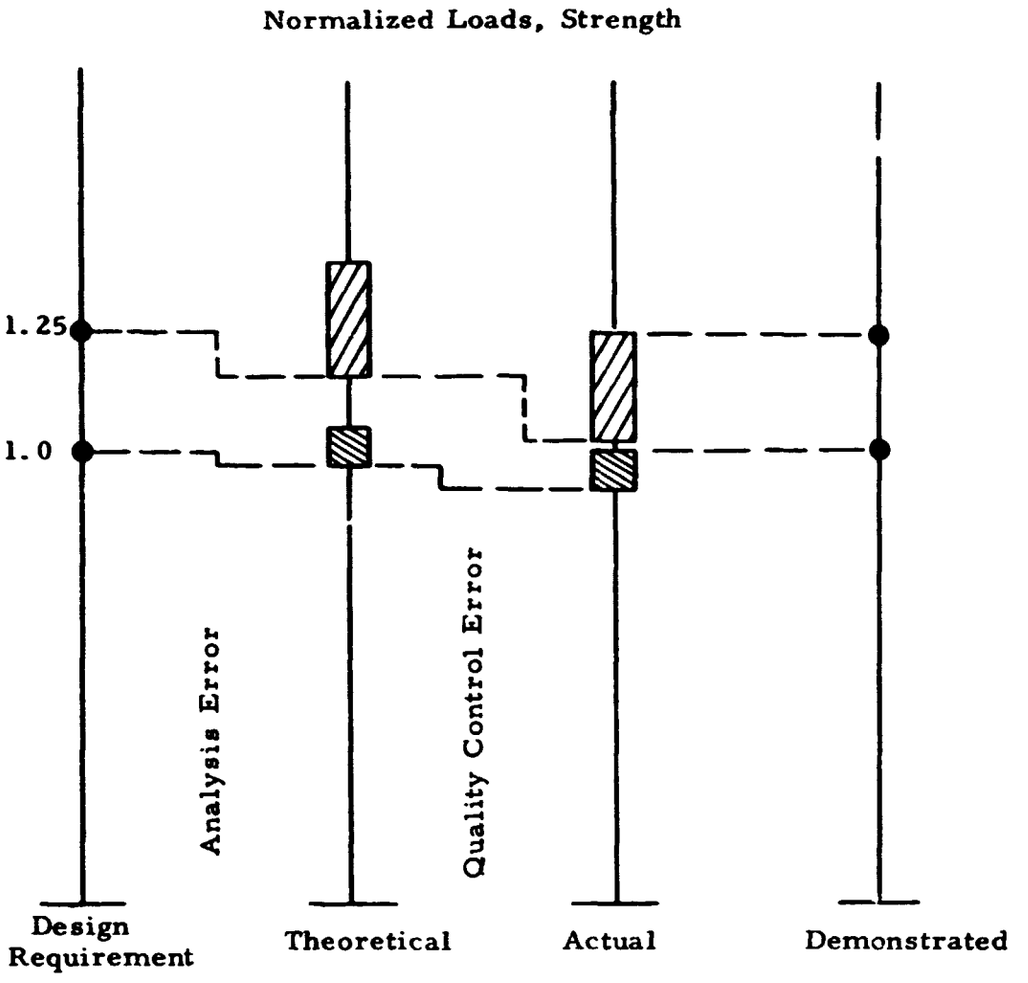


Figure A-2. Possible Lightweight, Test-Qualified Design/ Safety Factor Relationships

degradation may result in varying degrees in critical flight structure alignment. There will exist an unknown, reduced element of safety. If even more serious errors exist in the analysis, this will become apparent through test failure, and corrective design modifications will be required. Finally, if more serious quality control lapses occur, it is possible to unknowingly arrive at a state shown in Figure A-3 where hazardous flight failures will occur.

There are many other considerations in the actual development and qualification of flight hardware such as other environmental effects, life requirements, component tests, etc. It is believed, however, that the following general conclusions may be drawn for weight-optimized structure:

a. Flight structure safety (the ability to sustain limit loads without rupture or collapse) is established by the design ultimate safety factor, prototype test to ultimate load, and quality control which insures that all structures are at least 80 percent as capable as the prototype.

b. Flight structure reliability (the ability to perform its intended function in the design environment) is a function of the prototype test, degree of conservatism or error in the structural analysis, and quality control level. Some degradation in performance may result from either unconservative analysis or inadequate quality control. Reliability is established only if all flight structure is the same as, or better than, the test article. Fortunately, variations in yield capability are usually much less than variations in ultimate capability. Most payloads are not affected by minor yielding of primary structure, and even alignment-critical payloads can tolerate local yielding with only minor effect on mission quality. It is for these reasons that relatively small yield factors of safety (1.0 to 1.1) are used. It is then concluded that reliability is established within tolerable limits by test and quality control. Conservative design analysis will sometimes compensate for poor quality and will enhance reliability at the expense of weight, but some conservatism cannot be specified quantitatively or evaluated, analysis is not a definable contributor to reliability. Unconservatism in the analysis may lower mission quality but not below acceptable limits. Analysis errors which would result in unacceptable performance are detected by prototype tests.

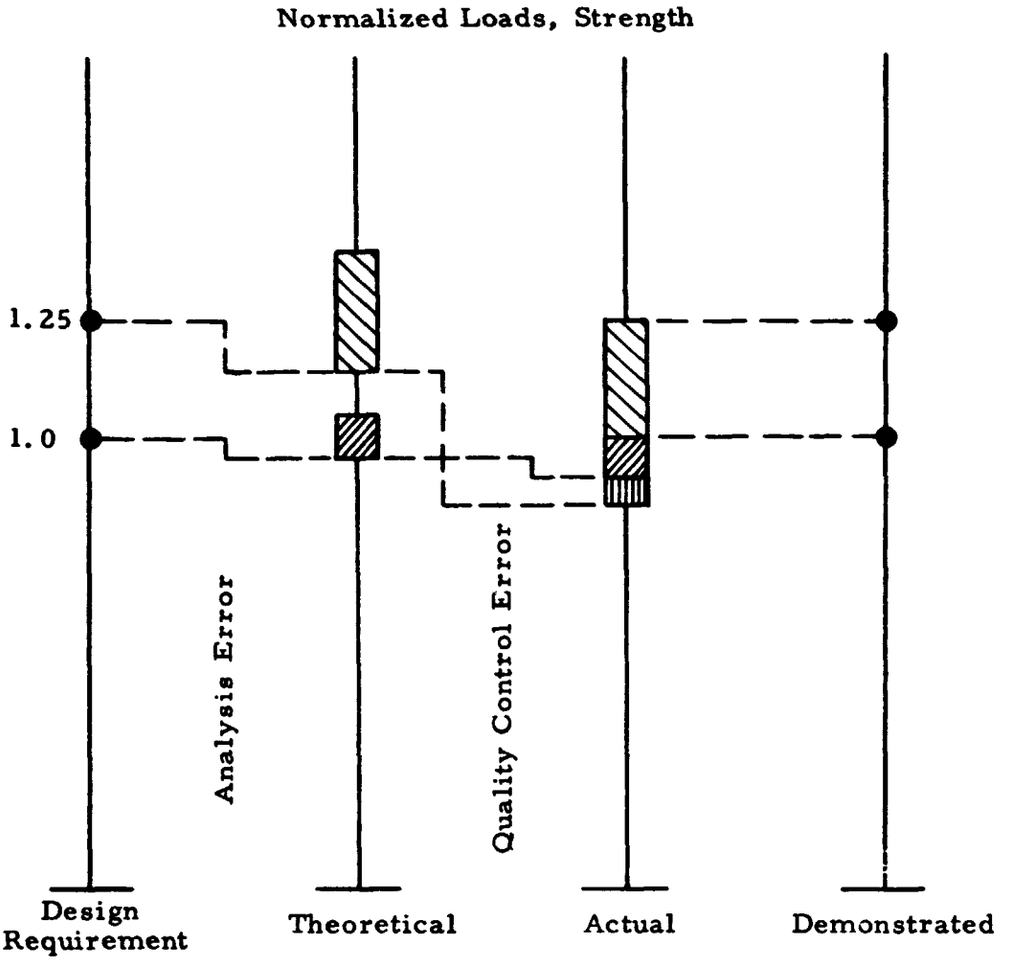


Figure A-3. Worst-Case Lightweight, Test-Qualified Design/ Safety Factor Relationships

c. Testing establishes the safety and reliability of the design.

d. Quality control establishes the safety and reliability of the untested flight hardware, and must be consistent with the safety factors utilized.

e. Analysis has only a minor and undeterminable effect on safety and reliability. Its primary function is to speed convergence of the design by eliminating excess weight and avoiding test failures, thereby reducing the development schedule. Analysis may be required for test planning and limit load determination, however, for complex structure governed by inertial response to transient events.

f. Accurate limit loads are essential. There is no allowance in the development or qualification for the payload structure to sustain overloading.

g. Safety factors for lightweight structure contribute to safety but not reliability (since the yield factor is 1.0) and do not compensate for errors in the design analysis, limit load determination, or quality control.

h. Analysis errors lead to test failures.

i. Quality control errors lead to flight failures.

In general, then, it is seen that a thorough test program and rigorous quality control are essential to hardware qualification. Most organizations also prepare more or less rigorous design analyses to prevent schedule problems and possibly for other reasons not directly related to schedule or quality control. As safety factors and weight are increased, potential reductions in these three areas of effort may be examined: analysis, test, and quality control. There are probably optimum values of design safety factor and upper bounds certainly exist, and it is these values and the corresponding requirements which are of interest.

5. INCREASED SAFETY FACTORS

If safety factors are increased above the values currently utilized for lightweight expendable payloads and other development and qualification procedures remain unchanged, there will be an increase in weight and in associated cost for the extra material. Any increase in strength will result in improved safety and reliability (the structure will be safe and reliable, for design loads as well as loading above limit load) or, alternatively, will

allow larger strength deviations between the test and flight structures. If it is assumed that limit loads for the Shuttle are adequately known by whatever means are required, there is no particular need for an improvement in safety and reliability. The possibility then exists of reducing the effort normally devoted to controlling strength scatter; namely, the specification of close drawing tolerances on dimensions and straightness and rigid specifications on procurement, process control, and inspection. Lumping this effort into the category of quality control, Figure A-4 shows the relationship between prototype-demonstrated safety factor and allowable strength scatter to prevent failure of untested flight hardware at limit load.

Figure A-4 also shows that considerable reduction in quality control effort can be obtained with modest weight increases. For example, an ultimate factor of safety of 2 increases the structure weight by less than 60 percent (going from F.S. = 1.25 to 2.00) and the satellite weight by less than 12 percent (depending on structure type) while increasing the the allowable strength scatter by 150 percent (from 0.2 to 0.5). It would appear that assuring quality of only 50 percent of perfect would involve very minimal effort compared to that currently exerted to obtain 80 percent quality.

The curve in Figure A-4 also applies to yield factors of safety, except that control of yield or deflection requires even less quality control. Prototype test-qualified structures thus appear to offer a promising tradeoff between weight and quality control effort. This is a trade which should be considered by the spacecraft contractor, based on his knowledge of current costs and consideration of alternate techniques which will assure quality, but within larger bounds.

In addition to increased weight and potentially lower quality control costs, there will be other associated changes which also require consideration. Test activity will remain the same as for lightweight structure except that larger loads, consistent with the higher safety factors, must be applied. Design analysis, for whatever purpose it is normally done, is not increased

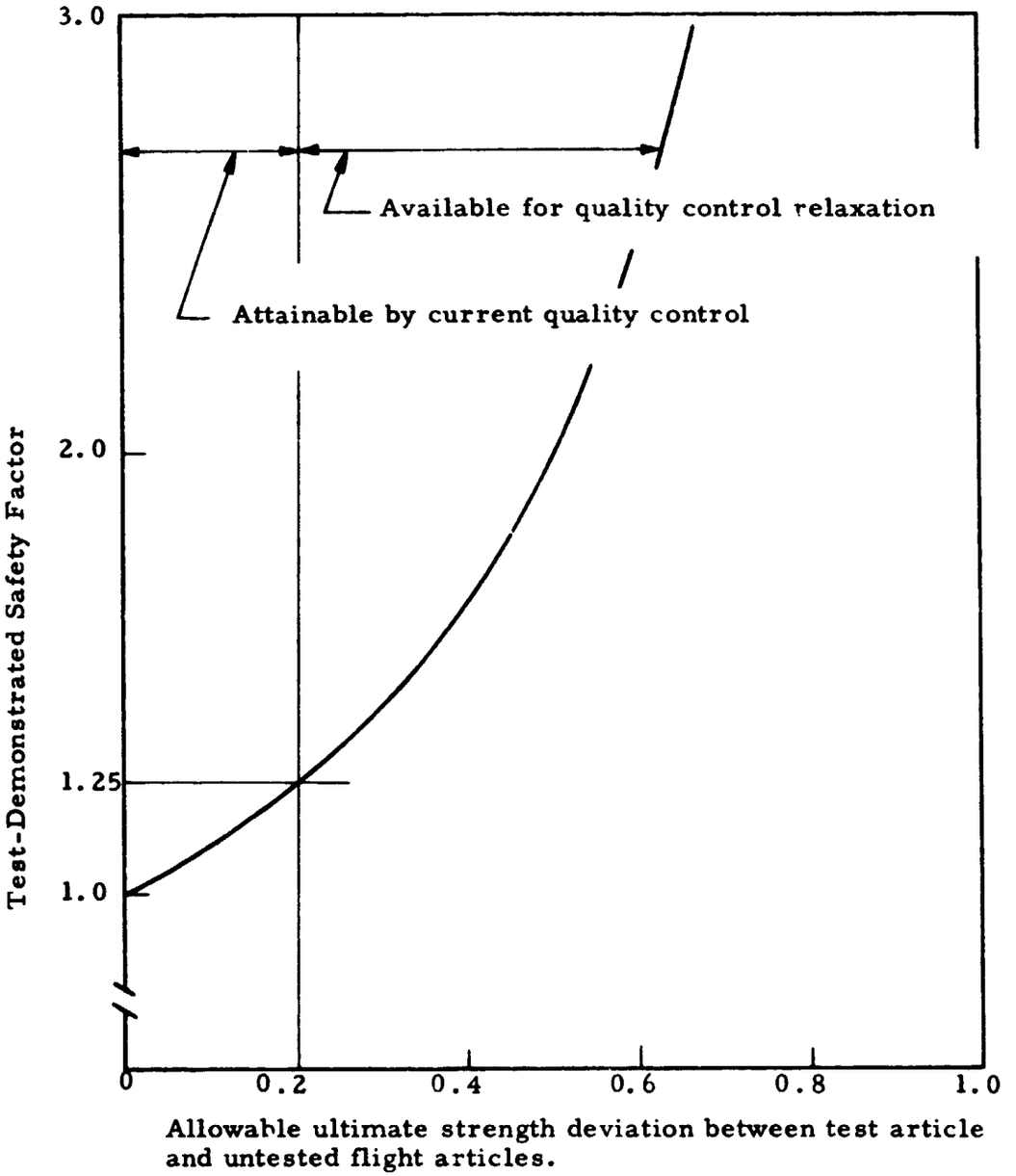


Figure A-4. Prototype Test-Qualified Structure

or decreased in scope. There will be a non-recurring effort required for each contractor to determine new procedures in procurement and inspection to provide consistent quality at a lower level. There will also be a slight increase in the cost of fabricating and joining heavier structure as well as a potential for nominal schedule improvement related to simplified procurement and inspection.

This approach does not imply that carelessness is now more tolerable. It merely allows simplified techniques by increasing acceptable limits on dimensions, material flaws, and workmanship quality. A side benefit of fewer rejected parts and less salvage activity will also be realized.

A second cost/weight trade becomes possible when both yield and ultimate factors are increased to a level somewhat above 1.25, if it is also required that no structural yield or damage result after application of test loads. The required design and test factors must be determined by consideration of cyclic loading and life requirements so that the test demonstrates a factor of safety of 1.25 after the test cycle. This approach will provide a step reduction in program cost by allowing flight of the test item.

Flight use of the test article will require that the test be sufficiently instrumented to detect any permanent reduction in capability. An increased dependence on the design analysis may be required to identify critical areas for instrumentation. Post-test inspection will be required, and special considerations are necessary so that loads may be applied without alteration of the structural design. Cost benefits will be approximately $45.00/kg ($20/lb) for fabricated aluminum payload structure, less any additional test and analysis cost.

These two approaches, reduced quality control and re-use of the test article, may be combined. Doing so suggests another possible economy: testing to a level above the 1.25 limit but below the design value. This will reduce test failures associated with design errors, a benefit whose worth must be evaluated by each particular design organization based on their test experience and design confidence.

One-of-a-kind payloads would seem to benefit most by such an under-test approach, particularly where development schedules are short. Structural safety and reliability for programs with a single payload are not determined by quality control or analysis but by direct demonstration.

In the preceding discussion, it is noted that confidence in the safety and reliability of the test article is higher than for untested structure by removal of the reliance on quality control. Therefore, the possible proof or acceptance testing of each structure should be explored. The large costs normally associated with qualification tests would almost certainly result in total cost increase unless ways can be found to simplify acceptance tests. It would seem that the most promising way of simplifying testing would involve changes to the basic structural design. If the structure is designed to an enveloping test condition rather than flight conditions, for example, it may be possible to adequately test some configurations by application of loads at a single point. The fitting to accommodate these loads could also serve as a convenient means of handling the payload, eliminating special handling fixtures.

6. UNTESTED STRUCTURE

The relationships shown in Figures A-1 through A-3 show clearly the demands which are placed on analytical effort in the absence of a structural test. For low factors of safety there is a dependence on analytical effort in the absence of a structural test. For low factors of safety there is a dependence on analytical and inspection thoroughness and accuracy which past test experience has shown to be unacceptable. Several authors have studied the error disclosure function of testing from a statistical approach and have concluded generally that is is impractical to increase safety factors to a level necessary to achieve acceptably low service failures without a demonstration of capability. Their results indicate a range in required safety factors of from about 4.0 to 11.0 (Ref. A-1) with some authors taking the position that only an infinite factor would suffice. Any of these values are clearly either impossible or economically infeasible for complete

aircraft or aerospace systems of current technology, except possibly in special applications which are not constrained by weight. Nevertheless, an attempt will still be made to define no-test safety factors for Shuttle-launched payloads through an examination of mitigating Shuttle considerations and attaining contemporary deterministic, rather than 3σ, levels of performance and safety.

Re-examination of Figure A-4 suggests that no-test safety factors may be determined if all of the uncertainty between predicted strength and actual strength is known. This will involve the usual quality control tolerance band in addition to analytically based uncertainty, which consists of:

a. Uncertainty of detailed stresses

b. Uncertainty of allowable stresses

c. Numerical error.

If these cannot be adequately bounded, tests will be required.

Uncertainty of detailed stress distribution is a function of structural configuration, and may be controlled to some extent by the analytical rigor applied. Uncertainty associated with allowable stresses is also related to structural complexity, depending on knowledge of boundary conditions, failure modes, and applicability of available empirical data. This indicates that design factor of safety requirements for untested designs should depend primarily on the relative complexity of the structural forms and assemblages utilized. Recommended safety factors for various test options are given in Table A-1 and reflected in the design criteria in Appendix B. The complexity ratings shown are subjective and must ultimately be negotiated with the procuring agency responsible for structure qualification. A general guide and examples of varying degrees of complexity are given in Appendix C for this purpose. The general philosophy allows the use of untested structure designed to increase safety factors if the structure is simple (amenable to conservative analysis) and if adequate analysis is performed. For more complex structural forms, higher factors are required together with more comprehensive analysis. There are some very complex structure types

Table A-1. Design Factors of Safety and Test Options[1]

	Yield	Ultimate
Qualification-Tested Designs		
Dedicated Test Article[2]	1.00	1.25
Flight Sample Test Article[3]	1.25 min	1.25 min
Acceptance-Tested Designs[4]	1.10 min	1.10 min
Development-Tested Structural Subassembly		
Dedicated Test Article	1.00	1.25
Flight Sample Test Article[3]	1.25 min	1.25 min
Untested Designs or Subassemblies[5]		
I Simple Structure	1.5	2.0
II	1.9	2.5
III	2.3	3.0
IV	2.7	3.5
V Moderately Complex Structure	3.0	4.0
VI Complex Structure or Uncertain Analysis	Test Required	Test Required

NOTES:

1. All factors presume rigorous quality control (80 percent of ideal). Relaxation of quality requires larger safety factors.
2. Qualification testing implies the flight of at least one untested article. Test loads are the same as design loads.
3. Test loads must be determined from life requirements.
4. Acceptance testing is performed on each flight article. Test loads must be determined from life requirements.
5. Untested structure requires rigorous conservative stress analysis. Complexity rating examples are given in Appendix C.

which will require testing, in which case design safety factors are determined from other considerations previously discussed. Simplicity of design thus affords cost-saving potential in development, fabrication, and the elimination of testing.

The recommended design safety factors shown in Table A-1 are somewhat arbitrary by necessity. Tested design factors are consistent with current DOD practice which, together with the associated surveillance and analysis requirements imposed on contractors, has produced adequate structure. The no-test factors acknowledge that there will be some degree of over-estimation of load-carrying capability, even with high analysis standards and independent review. It is felt, however, that the factors proposed are sufficient to cover these variations if current analysis standards are maintained and an objective review procedure established.

Referring again to Figure A-4, it is seen that with current quality control techniques, there will exist an additional allowance for analytical variations which range from about 25 percent underprediction for the simplest structure to about 55 percent underprediction for the most complex structure for which analytical qualification is allowed. A quality control range of 20 percent also exists.

A search for compiled static test experience has produced only one such data set. Failure levels for 450 static tests of military aircraft wings, fuselages, landing gear, and horizontal and vertical stabilizers during the 1940's have been compiled. There were 44 failures below ultimate load of which 3 were more than 50 percent understrength. Two were 60 percent understrength, and one was 65 percent understrength. The compilation does not separate the contributions of analysis and quality control in these failures (Ref. A-2).

None of these structures would have experienced service failures if designed to the recommended factors shown in Table A-1. Additional confidence in the recommended approach is gained by consideration of advances in analysis technology since that time and the relatively benign loads environment associated with Shuttle missions.

The increased importance of analysis for untested structure is reflected in the general analysis requirements below. Control of numerical errors is best accomplished by an independent review or evaluation of the analysis. It should be required that this objective surveillance be performed by a government organization or their specified representative.

7. ANALYSIS REQUIREMENTS FOR UNTESTED DESIGNS

Structural design is strongly affected by objectives which cannot be absolutely defined or imposed because of performance, risk, cost, or other factors that depend on hazards involved, the mission constraints, or details of the vehicle, its structure, and mode of operation. The primary objectives addressed here are the requirements for the design of a Space Shuttle payload which eliminates or minimizes proof of performance testing. Weight and volume constraints are considered to be of secondary importance. However, the payload must safely complete the required number of missions in its specified life and accomplish this at minimum cost. The following paragraphs present the requirements which must be met in order to achieve the design objectives.

The spacecraft design shall possess sufficient strength, rigidity, and other necessary characteristics that are required to survive the critical loading conditions that exist within the envelope of mission requirements as dictated by all life phases. It shall survive those conditions in a manner that does not reduce the probability of the successful completion of the mission. The structure shall not be designed to withstand loads, pressures, or environments due to malfunctions that would create conditions outside the expected mission design envelope.

Rigor and conservatism shall be exercised in the construction and use of mathematical structural models that describe spacecraft performance. The models shall be sufficiently comprehensive and empirically well founded to assure the accurate prediction of all pertinent aspects of structural behavior.

In the formulation of the mathematical model, the magnitude, direction, spatial distribution, and time variation of all external limit mechanical loads, pressures, and forcing functions shall be determined by conservative analyses of the design environments unless provided by NASA in the form of enveloping quasi-static loads. In addition, the temperature distribution throughout the spacecraft and its time dependence shall be accurately determined by suitable analyses of the design environments. The constitutive relations which accurately characterize the relation between material stresses, strains, temperature, and time shall be based on extensive test data which accounts for the possible influences of specimen-to-specimen scatter, material behavior nonlinearities, stiffness eccentricity effects, and degrading environmental effects. The mathematical model shall properly incorporate the appropriate boundary, juncture, and initial conditions. When uncertainty arises, these conditions shall be specified in various candidate models so that pertinent aspects of structural behavior can be conservatively bounded. The mass distribution, damping factors, and other quantities of importance to the dynamic behavior of the spacecraft shall be accurately determined and incorporated into the mathematical model. If large structural displacements appear likely, the equations of motion shall include appropriate geometric nonlinearities.

The analyst shall carefully configure the mathematical models to insure that all redundant load paths, cutouts, sources of prestress, stress concentration, structural misalignments, and critical dimensional tolerances have been accurately accounted for. In addition, sources of flexibility/stiffness due to flexible payload interfaces, bearing systems, mechanisms, and primary load-path subsystems such as stowed or deployed antennas and solar arrays shall be determined by appropriate analyses or subsystem development tests and accounted for in the payload mathematical models.

When uncertainty exists in regard to the assumptions made in the formulation of spacecraft mathematical models, the extent of the uncertainty

should be established by candidate formulations which conservatively bound the extent of the uncertainty or by low-cost subsystem development tests. Carefully administered general systems engineering/technical direction, supported by an independent analysis, is highly recommended for spacecraft sytems that are qualified by analysis only. The structural design should emphasize simplicity in form and combination of components, assemblies, and systems to facilitate accurate determination of load paths and to increase confidence in the calculated structural behavior. Construction features, for which test data and theory are in substantial agreements, should be utilized to the fullest extent possible.

Values for the allowable mechanical properties of the spacecraft structure in its design environment shall be taken from sources approved by NASA, such as MIL-HDBK-5B, MIL-HDBK-17, MIL-HDBK-23A, or the AFML Advanced Composite Design Guide. When values for mechanical properties of materials or joints are not available because they are new or used in a new environment, they shall be determined by analytical or test methods approved by NASA. A sufficient number of tests shall be performed to establish values for the mechanical properties on a statistical basis. The tests shall be performed in accordance with procedures in ASTM E8-69, E9-70, and AFML-TR-66-386. "A" values (99 percent nonexceedance with 95 percent confidence) shall be obtained and used for allowable stresses. The effects of temperature, thermal cycling, heating rate, processing variables, and detrimental environments shall be accounted for in defining allowable mechanical properties. Fatigue-life characteristics of the payload structure shall be obtained from references approved by NASA, or by experiment, for appropriate cyclical loading and loading rates in the applicable environments. Characteristics of the payload structural materials as regards failure due to cracking, corrosion, creep, meteoroid impact, or irradiation shall be similarly established so that these failure mechanisms can be avoided during the service life of the payload.

The mathematical model shall yield time histories. as appropriate, of pertinent internal displacements, rotations, stresses, strains, loads, and moments in terms of the applied loads and temperatures. In addition, the model shall predict susceptibility of the payload to general instability buckling, panel buckling, and crippling. These theoretical values shall be reduced to allowable values by empirically determined conservative knock-down factors, as approved by NASA, which account for the influence of characteristic structural imperfections. In this regard, historical buckling failure data should be utilized to the fullest possible extent.

Based upon a conservative evaluation of the internal loads time histories, the payload shall be designed to possess adequate strength, stiffness, and other necessary characteristics to withstand limit loads in the expected operating environments throughout its service life without experiencing detrimental deformation. Similarly, the payload shall be designed to withstand ultimate loads in the expected operating environments without experiencing collapse or rupture. Design factors shall be applied to limit loads but not to the stresses arising from temperature differences and gradients. The margin of safety shall be positive and shall be determined at allowable ultimate and yield levels and at the temperatures expected for all critical conditions.

8. QUALITY CONTROL REQUIREMENTS

The control of quality involves the coordinated efforts of many scientific and production disciplines and cannot be treated exhaustively here. The importance of quality is obvious, not for the attainment of perfection but in the ability to rely on lower-bound quality levels. Through considerable experience in aircraft and lightweight aerospace structures, quality standards and procedures have evolved which are consistent with qualification and acceptance methods. Combined with the traditional design factors, acceptable reliability and safety have resulted.

Continuation of these quality-screening procedures is necessary for untested low-cost structure. Acceptance testing is actually a quality-limiting process in itself which requires that any contemplated changes

in procurement, inspection, or fabrication quality control be based solely on a willingness to experience more, or less, acceptance test failures. This is the case regardless of whether the structure is weight or cost optimized. Over-designed structure, however, offers the possibility of designing to factors higher than those shown in Table A-1 for acceptance-tested structure so that quality standards may be relaxed without incurring higher test failure rates. This type of trade does not affect safety or reliability at all; it is a direct trade of quality control cost against structure weight and cost and the cost of test failures.

Qualification-verified designs also allow relaxed quality standards, but since quality in this case is an essential safety and reliability determiner, it is necessary to establish new standards of control which will essentially guarantee lower-bound quality at a new level. This is a task which must be carried out as a separate effort, preferably on contractors.

9. EXPENDABLE PAYLOADS

The discussion and criteria developed herein apply to all mission types. The design of expendable mission payloads will not differ from the other three mission types, because the payload may be returned if the payload does not satisfactorily pass the on-orbit predeployment checkout.

10. ON-ORBIT MAINTAINABLE PAYLOADS

No special considerations are necessary for on-orbit maintainable payload missions, since it is a basic premise that structure should be designed to include the necessity of on-orbit maintenance.

11. GROUND-REFURBISHABLE PAYLOADS

All structure should be designed to the specified number of missions and lifetime. Special situations may develop, however, where it may be advantageous to design to shorter life and reverify further use by periodic acceptance tests. This technique may also be useful in extending the mission cycles beyond original design. In any case, the structure should be examined for evidence of distress in critical areas and checked for alignment at each ground-refurbishment phase.

12. SORTIE PAYLOADS

As with all man-tended missions, it is possible to incorporate into the design provisions for real-time vernier alignment of components. While this would tend to reduce the required design yield factor of safety, it is recommended that the criteria in Appendix B be applied uniformly to all payload types.

C. CONCLUSIONS

It is concluded that the attainment of near-optimum payload structure, considered from a total development and production cost standpoint, requires that the potential economies of over-designed structure be considered. Several possible tradeoffs are identified in this appendix between structural weight and the cost-related activities associated with development, fabrication, analysis, test, and quality control. When these are considered in relation to a specific contractor's experience, specialty field, test facilities, and cost experience, it will be possible to select the most promising trades for detailed cost studies. It is also intended that some of the approaches discussed in this report, being somewhat unconventional, will inspire further re-examination of current traditions and expose additional cost-saving ideas. A general conclusion, further reinforced by this study, is that simplicity of design and qualification is the single most significant cost reducer in the absence of weight constraints, when properly executed.

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APPENDIX B. LOW-COST SPACECRAFT STRUCTURAL DESIGN CRITERIA GUIDELINES FOR SHUTTLE-LAUNCHED PAYLOADS*

A. INTRODUCTION

1. SCOPE

This appendix presents the basic requirements and information governing the design of the spacecraft structural subsystem for Space Shuttle payloads. Included herein are:

a. Basic facts and references pertinent to the structural design

b. Basic objectives for the philosophy of structural design

c. Conditions and environments for which the vehicle must be investigated and designed

d. Requirements for establishing loads and other environmental factors for the design conditions

e. Requirements and alternatives for establishing evidence of structural adequacy

f. Definition of terms.

2. AUTHORITY

This document shall govern the design of the spacecraft structural subsystem. Any deviation from these requirements shall be noted along with proper justification.

3. INTENT

The intent of these criteria is to provide a set of design conditions, requirements, and objectives which, when implemented, will insure that the structural subsystem achieves acceptable and compatible structural integrity.

*This appendix is based on work performed and published in an Aerospace Corporation internal report by J. E. Anderson and J. F. Hook, Spacecraft Structural Design Criteria Document, ATM-68(3305)-61 (June 1968).

B. DEFINITIONS, NOMENCLATURE, AND REFERENCES

1. DEFINITIONS

a. Limit Load

Limit load is the maximum anticipated load, or critical combination of loads, which a structure may be expected to experience during the performance of specified missions in specified environments. Since the actual loads that are experienced in service are in part random in nature, statistical methods for predicting limit loads shall be employed wherever appropriate.

b. Yield Load

Yield load is obtained by multiplying the limit load by the yield factor of safety.

c. Ultimate Load

Ultimate load is obtained my multiplying the limit load by the ultimate factor of safety.

d. Factor of Safety

The factor of safety is an arbitrary factor meant to account for uncertainties and variations from item to item in material properties, fabrication quality and details, and internal loads distribution.

e. Applied Temperatures

Applied temperatures are maximum calculated temperatures to which the structure will be subjected during the performance of specified missions in specified environments.

f. Critical Condition

A critical condition is a loading or temperature condition which dictates the design of a portion of the structure.

g. Failure

A structure is considered to have failed when it can no longer perform its intended function. Failure of a structure has occurred when a condition exists which may result in the loss of the vehicle, or any part thereof, and/or may present a hazard to operating personnel.

2. REFERENCES/APPLICABLE DOCUMENTS

The following documents are applicable to the extent specified herein.

a. MIL-HDBK 5B, "Strength of Metal Aircraft Element"

b. MIL-HDBK 17, "Plastics for Flight Vehicles"

c. MIL-M-8555A, "Missiles, Guided: Design and Construction, General Specifications for"

d. Shuttle Environmental Requirements Specification

e. Shuttle Loads Specification

f. Meteoroid Environment and Penetration Criterion

When a conflict occurs between the applicable documents and this document, the latter shall govern.

C. DESIGN CRITERIA

1. GENERAL DESIGN PHILOSOPHY

The structure shall possess sufficient strength, rigidity, and other necessary characteristics required to survive the critical loading conditions that exist within the envelope of mission requirements as dictated by all life phases. It shall survive those conditions in a manner that does not reduce the probability of a successful completion of the mission.

Consistent with the structural design principles and assumptions listed herein, the structure shall be designed to achieve minimum cost wherever practicable. Proper consideration shall be given to the effect on system weight and development schedule.

The structure shall be designed by consideration of all critical flight and non-flight conditions. Provisions shall be incorporated in the design for assembling, handling, transporting, and storing which will minimize overall mission costs.

2. DESIGN CONDITIONS AND ENVIRONMENTS

The environmental phenomena corresponding to each design condition shall include factors that can influence the structural design and typically include heating, vibration, shock, and acoustics in addition to quasi-static and dynamic loads. Consideration shall be given to the deteriorating effect of prolonged exposure to the space environment. Where possible, all such phenomena shall be determined statistically.

a. External and Internal Load Distribution

External loads shall be determined by conservative analyses of the design environment unless otherwise specified. The internal structural load distribution shall be determined by rational analyses. Effects of deformations, nonlinearities, and temperatures on internal loads distribution shall be included in analyzing the load distributions.

b. Combined Loads and Internal Pressure

When internal pressure effects in a combined load condition are stabilizing or otherwise beneficial to structural load capability, the minimum anticipated internal operating pressure for that condition shall be used instead of the ultimate design internal pressure in the ultimate loads analysis.

c. Malfunctions

The structure shall not be designed to withstand loads, pressures, or environments due to malfunctions that would create conditions outside the expected mission design envelope. For Shuttle mission phases, malfunctions shall not result in structural failures that jeopardize Shuttle or crew safety.

d. Misalignment and Dimensional Tolerances

The effects of allowable structural misalignments, control misalignments, and other permissible and expected dimensional tolerances shall be considered in the analysis of all loads, load distributions, and structural adequacy.

e. Dynamic Loads

Dynamic loads shall be determined for quasi-static and transient phenomena expected in each design environment. The calculation of all dynamic loads shall include the effects of vehicle structural flexibilities and damping and coupling of structural dynamics with the control system and the external environment.

Iterations of the dynamic loads calculations shall be performed as necessary to reflect design changes and/or mathematical model refinements. The final set of dynamic loads shall be determined with the use of experimental values of dynamic characteristics as obtained from appropriate tests and modal surveys.

f. Load and Thermal Fatigue

The effects of repeated loads and evaluated temperature will be considered in the structural design. The design structural adequacy of the vehicle in flight shall not be impaired by fatigue damage resulting from exposure to non-flight and launch environments.

g. Vibrational and Acoustical Loadings

The effects of the vibrational and acoustical environments shall be accounted for in design wherever possible by rational analysis of the response of the dynamic system to the environment.

h. Creep Deformation

The effects of permanent creep deformation shall be considered by rational methods of analysis. Where not otherwise critical; i.e., creep buckling, etc., a permanent deformation of one percent shall be considered as the maximum permissible value.

i. Thermal Stresses

The effects of thermal stresses shall be combined with the appropriate load stresses when calculating required strength. Thermal stresses shall be based on the applied temperatures, heating rates, etc. and shall be considered limit values.

3. MATERIAL PROPERTIES AND ALLOWABLES

Values for the allowable mechanical properties of the spacecraft structure in its design environment shall be taken from sources approved by NASA, such as MIL-HDBK-5B, MIL-HDBK-18, MIL-HDBK-23A, or the AFML advanced Composite Design Guide. When values for mechanical properties of materials or joints are not available because they are new or will be employed in a new environment, they shall be determined by analytical or test methods approved by NASA. A sufficient number of tests shall be performed to establish values for the mechanical properties on a statistical basis. The tests shall be performed in accordance with procedures in ASTM-E8-69, E9-70, and AFML-TR-66-386. "A" values (99 percent non-exceedance with 95 percent confidence) shall be obtained and used for allowable stresses. The effects of temperature, thermal cycling, heating rate, processing variables, and detrimental environments shall be accounted for in defining allowable mechanical properties. Fatigue-life characteristics of the payload structure shall be obtained from references approved by NASA, or by experiment, for appropriate cyclical loading and loading rates in the applicable environments. Characteristics of the payload structural materials as regards failure due to cracking, corrosion, creep, meteoroid impact, or irradiation shall be similarly established so that these failure mechanisms can be avoided during the service life of the payload.

4. STRENGTH REQUIREMENTS

a. At Yield Load

The structure shall be designed to have sufficient strength to withstand simultaneously the yield loads, applied temperature, and other

accompanying environmental phenomena for each design condition without experiencing excessive elastic or plastic deformation.

b. At Ultimate Load

The structure shall be designed to withstand simultaneously the ultimate loads, applied temperature, and other accompanying environmental phenomena without failure. No factor of safety is applied to any environmental phenomena except loads.

c. Margin of Safety

Margin of safety is defined as:

$$MR = \frac{1}{R} - 1$$

where R is the ratio of applied load (or stress, when applicable) to the allowable load (or stress). In determining the factor R, the effects of combined loads or stresses (interaction) shall be included.

5. STIFFNESS REQUIREMENTS

a. Under Yield Loads

The structure shall not experience excessive deformations at yield loads and in the appropriate design environment.

b. Under Ultimate Loads

Structural deformations shall not precipitate structural failure during any design conditions and environment at loads less than ultimate loads.

c. Component Stiffness

The fundamental resonant frequency of any component weighing 22.7 kg (50 lb) or less shall be 50 cps or greater when mounted on its immediate support structure, unless its structural integrity under dynamic loads is demonstrated by analysis or test.

6. THERMAL REQUIREMENTS

The effects of temperature shall be considered in design of the vehicle. Thermal analysis shall be based on rational transient analysis of heat fluxes from heating, engine exhaust gas radiations, engine system and electronic equipment heat sources, and consideration of the heat sink effect of the mass of structure, fuel, and equipment.

Thermal effects on the structure, including heating rates, temperatures, thermal stresses and deformations, and mechanical and physical property changes, will be based on a critical design heating environment.

7. FACTORS OF SAFETY

The structural subsystem shall be designed to the following tabulated factors of safety, which allow for alternative methods of establishing design adequacy. The design verification method selected shall be based on cost considerations which satisfy the general design philosophy.

Complexity ratings for untested design are given as examples only. Selected ratings must be supported by a rationale for their selection, based on analytical confidence for the particular design, and are subject to approval by NASA. Structural assemblies with mixed factors of safety are acceptable if component parts are developed in accordance with the requirement below. Factors higher than required are allowed and may be necessary to minimize overall deployment cost.

	Yield	Ultimate
Qualification-Tested Designs[1]		
Dedicated Test Article[2]	1.00	1.25
Flight Sample Test Article[3]	1.25 min	1.25 min
Acceptance-Tested Designs[4]	1.10 min	1.10 min
Development-Tested Structural Subassembly		
Dedicated Test Article	1.00	1.25
Flight Sample Test Article[3]	1.25 min	1.25 min

	Yield	Ultimate
Untested Designs or Subassemblies[5]		
I Simple Structure	1.5	2.0
II	1.9	2.5
III	2.3	3.0
V	2.7	3.5
V Moderately Complex Structure	3.0	4.0
VI Complex Structure or Uncertain Analysis	Test Required	

NOTES:

1. All factors presume rigorous quality control (80 percent if ideal). Relaxation of quality requires larger safety factors.
2. Qualification testing implies the flight of at least one untested article. Test loads are the same as design loads.
3. Test loads must be determined from life requirements.
4. Acceptance testing is performed on each flight article. Test loads must be determined from life requirements.
5. Untested structure requires rigorous conservative stress analysis. Complexity rating examples are given in Appendix C.

8. DESIGN PHASES

The payload shall be capable of withstanding design load conditions as specified (Ref. B-1) and environments to which it is exposed as specified (Ref. B-2) in addition to the orbit phase specified herein. Components shall be designed for the most severe environmental conditions with consideration of both operational and nonoperational states.

a. Orbit Phase

The payload shall be designed for geophysical environments and loading conditions associated with orbital flights. The design of the vehicle and its parts shall be based on, but not limited to, consideration of the following conditions.

b. Maneuvering Loads

Loads resulting from maneuvers for changing orbits, station-keeping, attitude control, and rendezvous along critical trajectories using actual propulsion, guidance, and control systems shall be determined.

c. Artifical Gravity

Where required by the mission, the loads induced by artifically simulated gravity shall be determined including the effects of perturbations such as docking with other vehicles, movement of equipment, initiation and termination of artificial gravity, etc.

d. Meteoroid Environment

The payload will encounter a flux of meteoroids as it orbits the earth. The structure, pressurized volumes, thermal radiators, solar panels, etc. shall be designed to achieve the required probability that meteoroid penetrations will not impair their satisfactory functioning. The magnitude and characteristics of this flux as well as methods for calculating the probability of penetration shall be taken as specified in Reference B-3.

e. Radiation Environment

The effect of both natural and artificial radiation environment shall be considered in designing the structure, radiators, solar panels, etc. including not only the deterioration and induced radiation effects on the materials, but also the shielding that may be required for sensitive equipment. The radiation environment is specified in Reference B-2.

f. Other Environmental Factors

Other geophysical environments peculiar to space must be considered in the design of the payload. These include the effects of vacuum and sputtering due to impinging atoms and ions. Items peculiar to individual vehicles such as internal pressurization and the thermal effects due to internal heating, solar heating, and extreme cold due to ambient space temperature must also be considered.

D. REQUIRED ANALYSES

1. EXTERNAL LOADS ANALYSES

External loads shall be developed in accordance with procedures and for the conditions specified in Reference B-1. From these external loads shall be developed consistent and conservative combinations of critical loads for application in internal loads analyses and instructural tests where required.

2. THERMAL ANALYSES

Thermal analyses for the payload shall be provided which shall include correlation of predicted and experimental heat transfer data if available. The analyses shall be in sufficient detail to provide transient and steady-state external surface temperatures and transient and steady-state internal temperature distributions for all primary and secondary structures as required to determine critical design conditions. The analyses shall consider the thermal conductivity of materials and internal radiation and convective effects on the heat transfer system. Time-temperature distribution plots for all structural components determined to be critical shall be presented in graphical form. The analyses shall be in sufficient detail that the inboard operating environments may be determined for associated internal equipment and secondary structure.

3. INTERNAL LOADS AND STRESS ANALYSES

Internal loads and stress analyses of the structure shall be performed by appropriate and conservative methods. The analyses shall indicate anticipated modes of failure for primary structure and shall demonstrate positive margins of safety. Stress analyses shall include consideration of fatigue, creep rupture, corrosion, thermal and mechanical history, and other detrimental effects that may result from service through the several design phases. Thermally induced stresses shall be combined with the stresses due to external loads in the stress analyses.

The mathematical model shall yield time histories of pertinent internal displacements, rotations, stresses, strains, loads, and moments in terms of the applied loads and temperatures. In addition, the model shall predict susceptibility of the payload to general instability buckling, panel buckling, and crippling; these theoretical values shall be reduced to allowable values by empirically determined conservative knockdown factors, as approved by NASA, which account for the influence of characteristic structural imperfections. In this regard, historical buckling failure data should be utilized to the fullest possible extent.

Rigor and conservatism shall be exercised in the construction and use of mathematical structural models that describe spacecraft performance. The models shall be sufficiently comprehensive and empirically well founded to assure the accurate prediction of all pertinent aspects of structural behavior.

In the formulation of the mathematical model, the magnitude, direction, spatial distribution, and time variation of external limit mechanical loads, pressures, and forcing functions shall be determined by conservative analyses of the design environments. In addition, the temperature distribution throughout the spacecraft, and its time dependence, shall be accurately determined by suitable analyses of the design environments. The constitutive relations which accurately characterize the relation between material stresses, strains, temperature, and time shall be based on extensive test data which accounts for the possible influences of specimen-to specimen scatter, material behavior nonlinearities, stiffness eccentricity effects, and degrading environmental effects. The mathematical model shall properly incorporate the appropriate boundary, juncture, and initial conditions. When uncertainty arises, these conditions shall be specified in various candidate models so that pertinent aspects of structural behavior can be conservatively bounded. The mass distribution, damping factors, and other quantities of importance to the dynamic behavior of the

spacecraft shall be accurately determined and incorporated into the mathematical model If large structural displacements appear likely, the equations of motion shall include appropriate geometric nonlinearities.

The analyst shall carefully configure the mathematical models to insure that all redundant load paths, cutouts, sources of prestress, stress concentration, structural misalignments, and critical dimensional tolerances have been accurately accounted for. In addition, sources of flexibility/ stiffness due to flexible payload interfaces, bearing systems, mechanisms, and primary load path subsystems such as stowed/deployed antennas and solar arrays shall be determined by appropriate analyses/ subsystem development tests and accounted for in the payload mathematical models.

When uncertainty exists in regard to the assumptions made in the formulation of spacecraft mathematical models, the extent of the uncertainty should be established by candidate formulations which conservatively bound the extent of the uncertainty or by low-cost subsystem development tests. Carefully administered general systems engineering/technical direction, supported by an independent analysis, is highly recommended for spacecraft systems that are qualified by analysis only. The structural design should emphasize simplicity in form and combination of components, assemblies, and systems to facilitate accurate determination of load paths and to increase confidence in the calculated structural behavior. Construction features, for which test data and theory are in substantial agreement, should be utilized to the fullest extent possible.

4. DYNAMIC ANALYSES

Intensities of acceleration, shock, vibration, and acoustic noise may vary with location in the cargo bay and phase of the mission profile. The equipment and structure shall be designed to the critical environmental conditions which are established for each particular location. In addition, the expected ground handling and transportation conditions shall be taken into account.

Dynamic loads analyses shall be conducted in cases where rapidly applied or fluctuating loads give rise to dynamic response of the vehicle or its component parts. Transient loading events of the boost phase of flight shall be analyzed, if required, with full consideration of the dynamic interactions of the spacecraft and Shuttle. The interaction of control system dynamics and structural dynamics shall be considered.

5. ANALYSES RELATED TO ORBITAL PERFORMANCE

In the case of spacecraft for which mission performance is affected by structural deformations and dynamic response to mechanical excitation, analyses sufficient to identify and evaluate the effects of these phenomena shall be performed. Thermal deformations as well as the dynamic response to excitation by rotating machinery and other moving parts, rocket motor thrust, etc. shall be considered.

E. REQUIRED TESTS

1. STATIC TESTS

a. Development Tests

Development tests are tests performed to determine and evaluate design feasibility, functional parameters, technological data, packaging and fabrication techniques, and environmental limitations. This category of testing includes tests designed to (1) demonstrate that the design meets the specified requirements, (2) identify critical areas where design improvement may be required, (3) identify primary failure modes or critical environments, or (4) demonstrate that the probability of passing qualification is sufficient high to warrant commitment of equipment to the qualification test. These tests may be conducted to any hardware level and include compatibility or integration tests. Development testing required to meet program objectives shall be conducted.

b. Qualification Tests

Qualification tests are tests performed to demonstrate adequacy of the structural design to sustain critical design loads and environmental

conditions in the manner required. The structural configuration, materials, and manufacturing processes employed in the qualification test specimens shall be identical to those of the flight articles.

When environmental conditions cannot be properly conservatively simulated in the test, allowances for material properties, combined loading, and other missing effects shall be provided in the test procedure and loads. Where prior landing histories affect the structural adequacy of a test article, these shall be included in all test requirements. Adequate instrumentation is required in order to properly evaluate the results of these tests.

The test article shall support 100-percent yield loads with no permanent deformation and without experiencing structural member deformation that will cause interference with other structural members and/or nonstructural components. The test article shall support 100-percent ultimate loads without failure of the primary structure or any other structure that may cause failure of the mission. The support fixture to be used in structural testing of all vehicle system sections shall consist of an adequate portion of the adjacent structural section to provide boundary or support conditions which are identical to the support conditions to be realized in the flight article.

c. Acceptance Tests

Acceptance tests are tests performed on deliverable equipment to detect and eliminate articles of substandard manufacture or of defective workmanship and to demonstrate, within the scope of the tests, that the properties of the equipment are identical within acceptable limits to those of the qualification articles. Acceptance testing of structure may be waived if product quality can be assured through appropriate quality control and inspection procedures.

2. DYNAMIC TESTS

a. Ground Vibration Tests

Ground vibration tests shall be conducted as necessary on the spacecraft for the purpose of experimentally determining natural frequencies, mode shapes, and structural damping. The test article should be of prime quality structure, complete in every detail with all significant installed equipment. Mass simulations of unavailable components may be acceptable.

Spacecraft mounting, or support, fixtures shall provide boundary conditions compatible with the conditions being simulated and shall prevent significant dynamic interactions of the test article with the support structure.

It shall be the objective of the mode survey test to thoroughly define a complete set of modes in the frequency range of interest (e.g., 0-50 cps for loads determinations). This will permit the utilization of measured modes directly in the desired analysis task, if required.

b. Fatigue and Acoustical Tests

Fatigue tests shall be performed as necessary on critical structural specimens to demonstrate fatigue life under vibratory loads. Acoustical tests of critical structural specimens shall be conducted to demonstrate the ability of the vehicle structure to satisfactorily survive the acoustical environment.

3. TESTS RELATED TO ORBITAL PERFORMANCE

Where dynamic response characteristics are critical to the adequacy of orbital performance, vibration mode surveys of the spacecraft system, or subsystems, shall be conducted to verify the validity of the analytical model. Similarly, equipments which give rise to disturbances on orbit shall be subjected to test to verify the nature and magnitude of these effects.

Where thermal deformations are critical to the adequacy of orbital performance, tests to verify analytical predictions of these effects shall be conducted. Applied temperatures may be determined by conservative

analysis where feasible; otherwise, the structure shall be subjected to sources of thermal radiation simulating those to which it will be exposed in critical orbit conditions.

F. REQUIRED DOCUMENTATION

Documentation of the required analyses and tests specified herein shall be prepared and submitted to NASA as contractual items. Analysis reports shall contain an adequate description of the data, assumptions, and methods used as well as a summary and discussion of significant results. Test reports shall contain adequate descriptions of the test setup, instrumentation, and loads or other excitations applied. Raw test data as well as reduced data shall be presented. Where appropriate, analyses of experimental error shall be included.

G. REFERENCES

B-1. B. Allesina, and E. F. Styer, Studies of Cost Effective Structures Design for Future Space Systems - Summary, NASA CR-1068, (June 1968).

B-2. I. Bouton, M. Fish, and D. J. Trent, Quantitative Structural Design Criteria by Statistical Methods, Volume II, AFFDL-TR-67-107, (February 1968).

B-3. Structural Design Factors, (Review Draft), NASA SP-80XXX (February 1973).

APPENDIX C. STRUCTURE COMPLEXITY RATING

This appendix defines and describes the complexity categorization used in selecting the design factor of safety.

A. COMPLEXITY I

Structural assemblies in this category are typically simple with relatively few, well-defined elements. Internal load distributions are accurately known by application of static equilibrium principles or simple redundant structure techniques. Members are uniform or of regularly varying cross-section without significant stress concentrations. Failure modes are well defined and preclude general or local instabilities, whether elastic or inelastic, and fatigue. Allowable stresses and loads are obtained from published minimum properties of conventional materials or handbook solutions whose application is rigorous. Structural elements in this category include uniform compact struts, beams, and thin rings. Axisymmetric straight-meridian monocoque shells, shear panels, and plates subject to known uniform loads are included. Structural assemblies of these elements, when simply and continuously connected, are Complexity I if there are no non-linear stress considerations to and including ultimate load.

B. COMPLEXITY II

Complexity II structures are those for which the number and redundancy of elements requires finite element or continuum analysis for internal loads distribution, thereby precluding absolute conservatism at every point. Stress concentrations which are small in relation to member dimensions and do not produce overlapping stress fields are allowed if covered by empirically verified handbook cases. Failure modes of local instability with well-defined loads and boundary conditions are allowed if based on verified published crippling or panel buckling data. Tension field

and fatigue failure modes are not permitted. Ring-stiffened shells, deep beams, and ri gs are included in Category II in assemblies which provide reasonable certainty of load distribution across element boundaries.

C. COMPLEXITY III

Structure in this category includes line-generated axisymmetric shells with general instability failure modes if amenable to conservative specification of boundary conditions and supported by empirically based critical buckling data. Post-buckling capabilities are discounted except for semi-tension field beams elastically buckled. Stable axisymmetric shells subject to concentrated or rapidly varying tangential loads are included if supported by suitably refined finite element analysis. Semimonocoque shells not subject to general instability failure modes are similarly included.

D. COMPLEXITY IV

Structures of this category require exhaustive computer analysis as a result of complex load paths among a large number of structural elements. Doubly curved, stable, axisymmetric shells are included for situations where boundary conditions are continuous and well defined and for tangential loads developable in three or fewer harmonics. Fatigue failure modes are included only in areas of known uniform stresses in conventional materials designed to the lower-bound endurance limit. This category also includes semimonocoque shells in instability failure modes when suitable finite element models are established which account for element-to-element eccentricities, and conservative application of appropriate critical buckling data is possible.

E. COMPLEXITY V

Structure forms and assemblies in this category are the most complex types for which qualification may be established by analysis

without test. All previous structure types and corresponding failure modes are included, together with the additional complexities introduced by access openings, either framed or flanged, and the inclusion of eccentric, discrete mechanical interface attachments. These added effects are normally associated with shell-type primary structure and, although local in nature, often are the details which limit overall payload load-carrying capability. Assignment of these effects to this category should only be made when the construction details and analysis refinement allow the accurate establishment of redundant load paths and localized stresses and failure modes.

F. COMPLEXITY VI

Structural forms in this category require development, acceptance, or qualification test regardless of design safety factor. Configurations, complexities, and failure modes not specifically included in Categories I-V are relegated to Category VI which includes:

1. General shells
2. Fittings, unless consisting of previously described elements
3. Normal, non-uniform loads on shells
4. Fatigue failure modes at stress risers
5. High temperature effects which introduce material non-linearities
6. Geometric non-linearities
7. Mechanisms and moving mechanical assemblies
8. Anisotropic materials, except metallic honeycomb.

APPENDIX D. FACTOR OF FETY FOR PRESSURE VESSELS

A. INTRODUCTION

In this appendix, the principles of linear elastic fracture mechanics were applied to establish the appropriate factors of safety for design and proof tests of pressure vessels for the four types of Shuttle payloads: expendable, on-orbit maintainable, ground-refurbishable, and Sortie. Aluminum, titanium, and steel were represented among the six common alloys selected for this study. A brief description of these alloys and the rationale leading to their initial selection are summarized below, and the material strengths and fracture toughness are shown in Table D-1.

1. 2219-T851 ALUMINUM

The Al-Cu alloy 2219-T851 is readily weldable and useful for applications at a wide range of temperatures from -452°F (~4°K) to 600°F (~589°K). It is a relatively tough material, is highly resistant to stress-corrosion cracking, and has been used in critical cyrogenic applications as well as those requiring high strength and creep resistance at relatively high temperatures (400°F - 600°F).

2. 6061-T651 ALUMINUM

Alloy 6061-T651 is a readily weldable Al-Mg-Si alloy possessing high resistance to corrosion. It has been used in a wide range of applications, including cyrogenic applications requiring high toughness.

3. 5Al-2.5Sn TITANIUM

The all-alpha alloy Ti-5Al-2.5Sn has excellent weldability, toughness at low temperatures, and long-term elevated temperature strengths. It is not so readily formed into complex shapes as other

Table D-1. Material Characteristics

MATERIAL PROPERTIES	ALUMINUM		TITANIUM		STEEL	
	2219-T851	6061-T651	6Al-4V	5Al-2.5Sn	T-1	HP9Ni-4Co-0.25C
C (upper bound)						
• $(MN/m^2)^{-2}$/cycle	1100×10^{-11}	400×10^{-11}	70×10^{-11}	130×10^{-11}	45×10^{-11}	70×10^{-11}
• ksi^{-2}/cycle	523×10^{-9}	190×10^{-9}	33×10^{-9}	62×10^{-9}	21×10^{-9}	33×10^{-9}
F_{ty}						
• MN/m^2	393	248	841	758	814	1220
• ksi	57	36	122	110	118	177
F_{tu}						
• MN/m^2	427	290	938	814	889	1269
• ksi	62	42	136	118	129	184
K_{Ic}						
• $(MN/m^2)\sqrt{m}$	38	30	82	127	101	109
• $ksi\sqrt{in}$	35	27	75	116	92	99
REFERENCES						
• C (upper bound)	2	4	6	8	9	11
• K_{Ic}	3	5	7	8	10	12
• Strengths	1	1	6	8	10	12

alloys with similar room-temperature properties but far surpasses them in weldability. Most forming and welding operations must be followed by an annealing treatment to relieve residual stresses imposed by the prior operation. It is metallurgically stable at moderate elevated temperatures. The alloy has good oxidation resistance up to 1050°F (~840°K).

4. 6Al-4V TITANIUM

The alloy Ti-6Ai-4V can be used in either the annealed or solution-treated-plus aged (STA) condition and is weldable. Temperature use range is from -320°F (~77.6°K) to 750°F (~672°K). For maximum toughness, this alloy should be used in the annealed condition, and for maximum strength, the STA condition. Stress-relief annealing after welding is recommended. This alloy is resistant to oxidation at least to 1000°F (~811°K).

5. T-1 STEEL

USS "T-1" steel is a low carbon, quenched, and tempered constructional alloy steel combining toughness at low temperature, high strength at elevated temperatures, ease of fabrication, and excellent weldability. It is a proven engineering material for pressure vessels and has superior atmospheric corrosion resistance compared with structural carbon steel. It is suitable for many applications involving temperatures up to 900°F (~755°K). This alloy must be used in the heat-treated condition, but stress relieving of weldments made of T-1 steel is not generally considered necessary because such weldments are tough in the "as-welded" condition. However, under certain conditions in which stress corrosion is involved, stress relieving may be necessary. T-1 steel may be readily cold formed or hot formed provided the hot forming is done at temperatures below 1100°F (~866°K) in order to preserve the original mechanical properties and toughness.

6. HP9Ni-4Co-0.25C STEEL

The steel alloy HP9Ni-4Co-0.25C was developed specifically for high hardening and toughness with ease of fabrication. This alloy requires quenching and tempering and is readily weldable, even in the heat-treated condition, without detrimental effect. Since no preheat or postheat are required, welding during refurbishing in the field is possible. Fabrication procedures are generally the same as for the familiar SAE 4340 steel. It is possible to cold form HP9Ni-4Co-0.25C even after heat treating.

B. GUIDELINE DEVELOPMENT

1. ASSUMPTIONS

Certain assumptions on fracture data were invoked to simplify the transformation of this guideline into a practical solution. Foremost of these assumptions were the following:

a. Upper bound flaw growth rate for each alloy is used to generate conservative values of proof and design factors of safety.

b. Although the deep flaw magnification factor M_k increases from 1.0 to 1.6 as the flaw deepens with additional cycles, this factor is taken as a constant value of 1.6 to provide a conservative design criterion.

c. The maximum values of the stress intensity factor are limited to the threshold values to minimize the effects of sustained loads flaw growth.

d. The stress level during proof test will be limited to the uniaxial yield strength of the material.

e. The pressure vessel shall be designed to withstand proof pressure without yield or excessive deformation and shall withstand ultimate pressure without rupture or collapse.

2. CYCLIC FLAW GROWTH LAW

The equation proposed by Forman, Kearny, and Engle (Ref. D-1) relating the rate of crack propagation under cyclic loading to the stress intensity factor range ΔK is

$$\frac{da}{dN} = \frac{C(\Delta K)^n}{(1-R)K_c - \Delta K} \qquad (1)$$

where $\frac{da}{dN}$ = crack growth rate, m/cycle;

$\Delta K = K_{max} - K_{min}$;

= range of stress intensity factor, $(MN/m^2)\sqrt{m}$;

R = load ratio in a given cycle;

$= K_{min}/K_{max}$;

K_c = critical stress intensity factor, $(MN/m^2)\sqrt{m}$;

C, n } coefficients determined emerically to provide best fit to test data

By writing this equation in the form

$$\frac{d(a/Q)}{dN} = \frac{C(\Delta K_{Ii})^3}{(1-R)K_{Ic} - \Delta K_{Ii}} \qquad (2)$$

where (a/Q) is the normalized flaw depth, K_{Ic} is the critical stress intensity factor, and ΔK_{Ii} is the range of initial stress intensity factor, Pittit et al (Ref. D-2) found that the growth of part through cracks in Inconel 718 at cryogenic temperatures was adequately described. Equation (2) was also found by Au and Speare (Ref. D-3) to apply to the data trend of Pendelberry et al (Ref. D-4) involving cyclic flaw-growth data for 9Ni4Co (Martensitic and Bainitic), 300 M (0.45c and 0.39c), and D6AC for an air environment at room temperature. Accordingly, Equation (2) will be used in this study as the appropriate cyclic flaw-growth law for the six selected materials. The suitability of this equation is evidenced by an examination of Figure D-1 for 2219-T851 Aluminum, Figure D-2 for 6Al-4V Titanium, and Figure D-3 for HP9Ni-4Co-0.25C Steel. It is seen that the scatter is appreciable. The upper bound values of C for each of the six materials are tabulated in Table D-1.

3. EQUATION FOR PROOF FACTORS

An equation relating proof test factor and the number of cycles remaining for a flaw to grow from some initial size ($\sim K_{Ii}$) to the threshold

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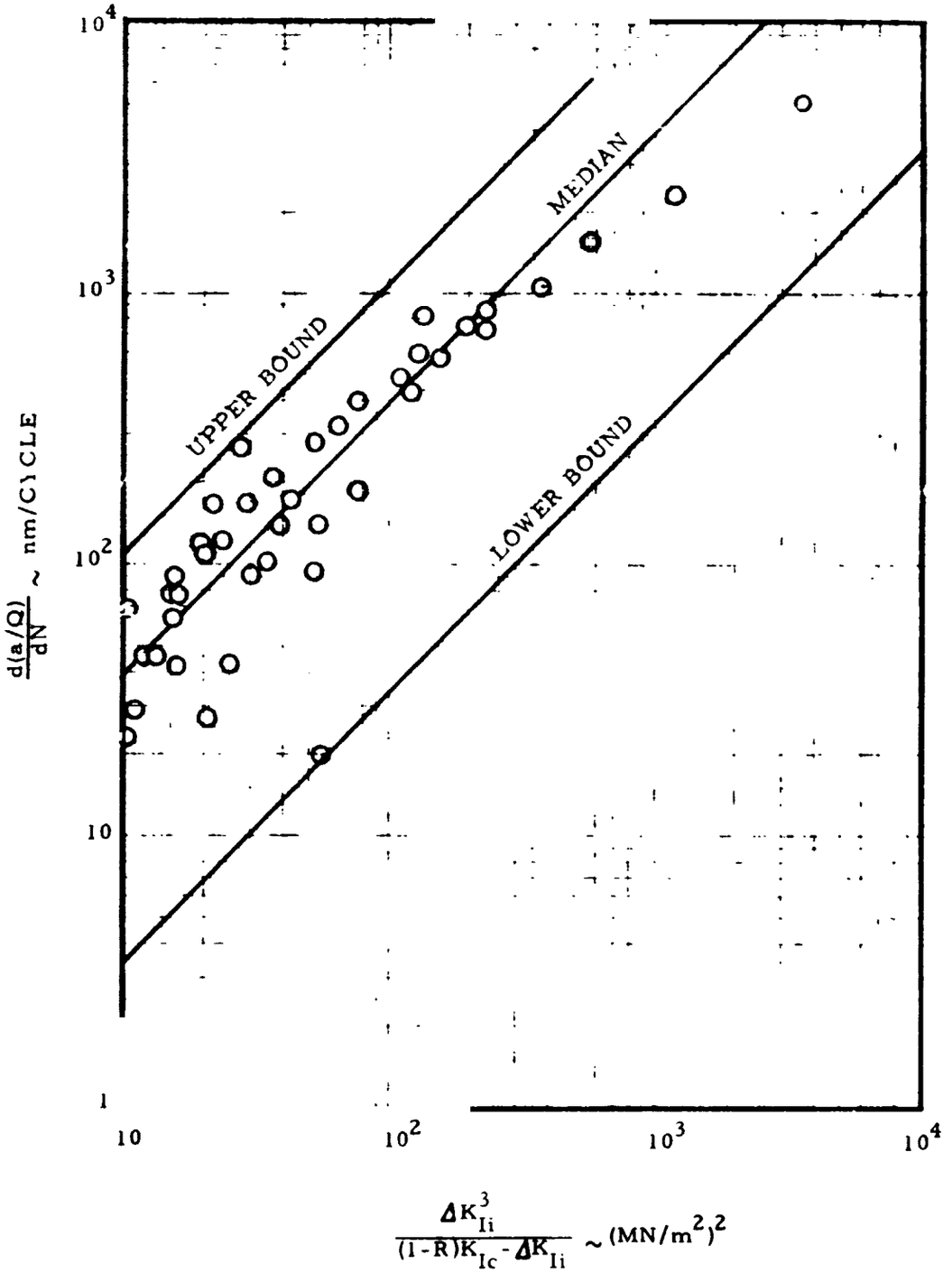


Figure D-1. Cyclic Flaw Growth Data for 2219-T851 Aluminum (Ref. D-5)

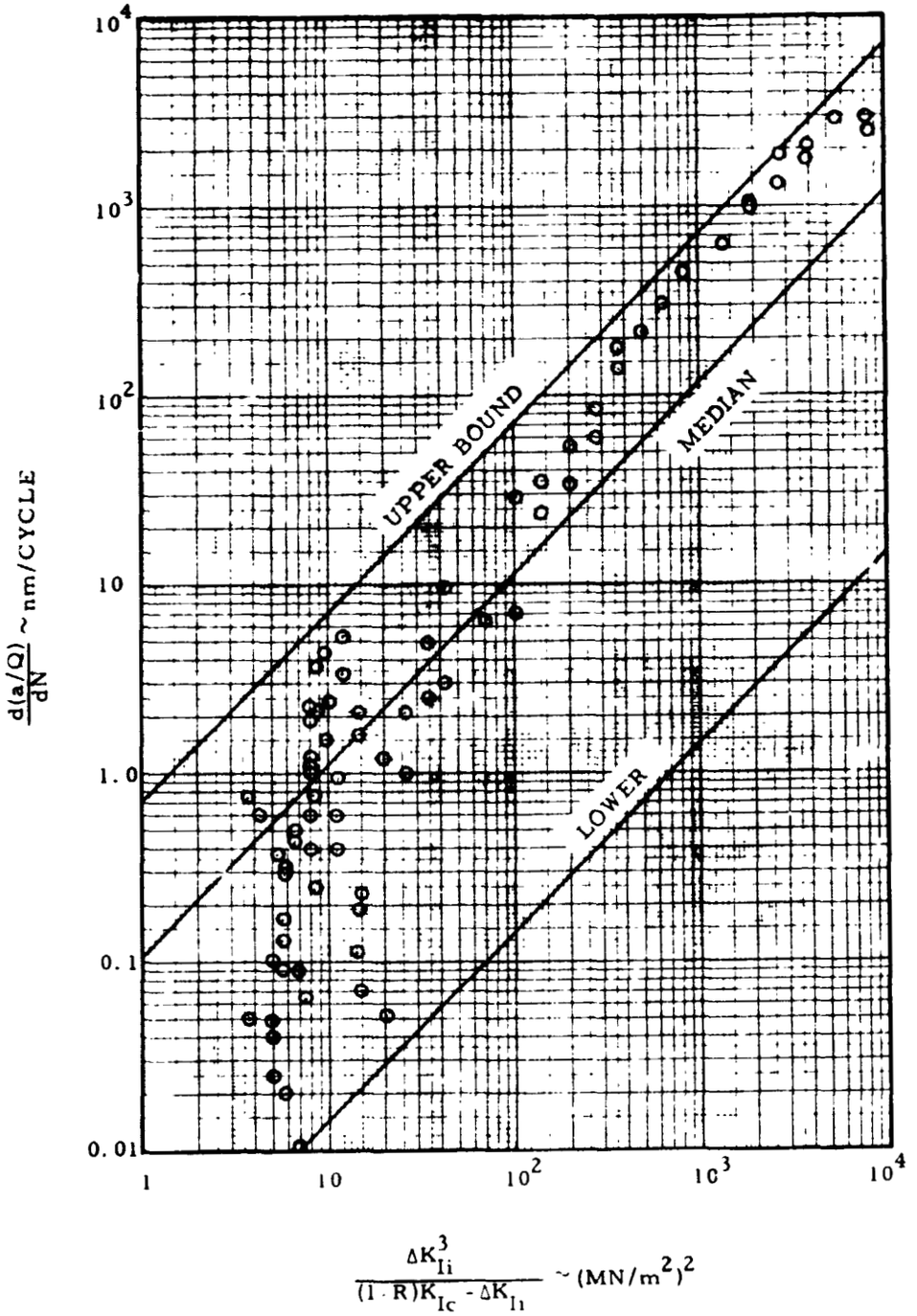


Figure D-2. Cyclic Flaw Growth Data for (Al-4V Titanium (Ref. D-6)

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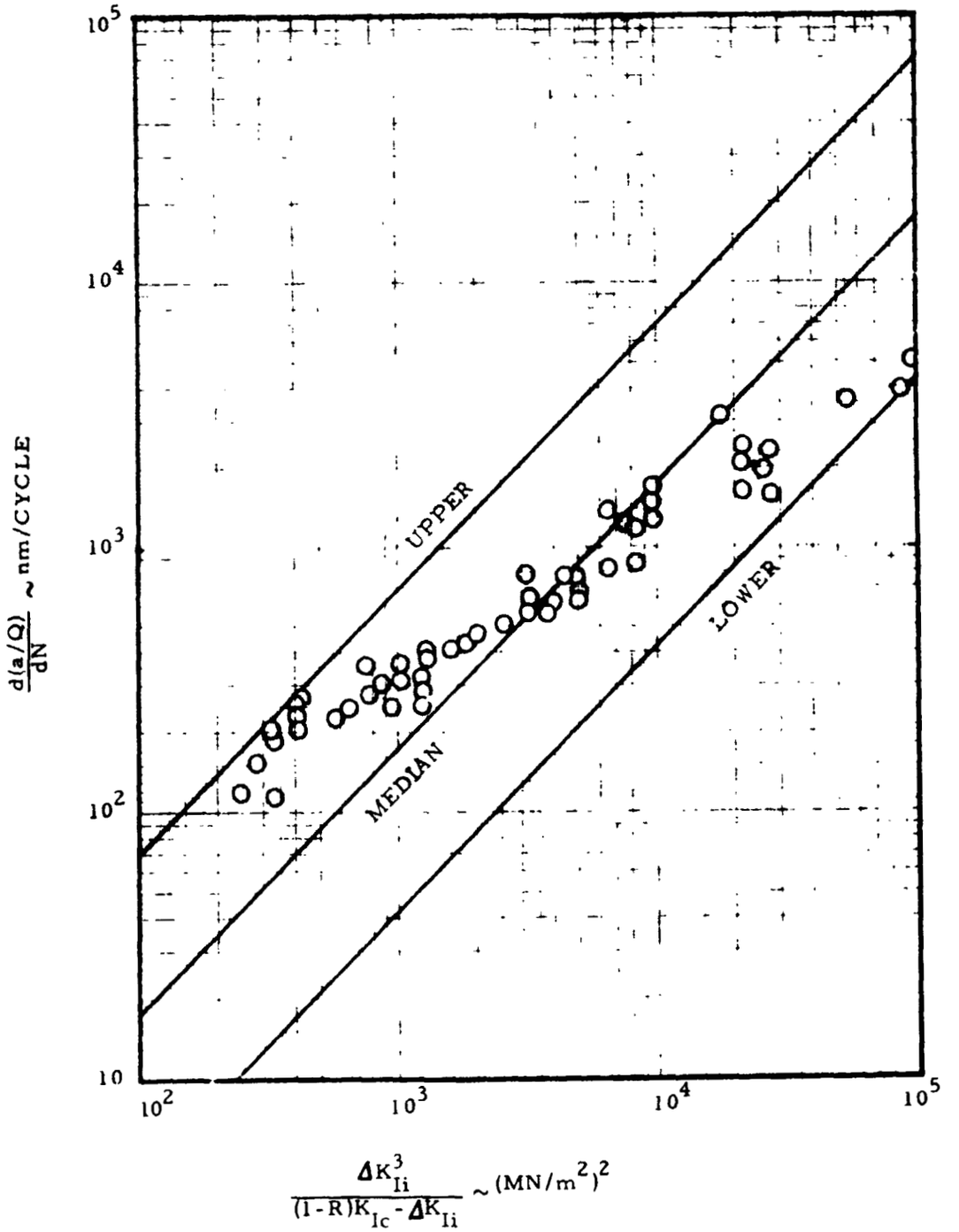


Figure D-3. Cyclic Flaw Growth Data for HP9Ni-4Co-0.25C Steel (Ref. D-7)

size ($\sim K_{TH}$) was previously derived by Au and Speare (Ref. D-3) using the conventional equations of linear elastic fracture mechanics and Equation (2). This equation is

$$N_{R(i\rightarrow TH)} = \frac{2\alpha^2}{C^* \sigma_{proof}^2} \left[\alpha - \frac{K_{Ic}}{K_{TH}} + \ell n \left(\frac{1}{\alpha} \frac{K_{Ic}}{K_{TH}} \right) \right] \qquad (3)$$

where $N_{R(i\rightarrow TH)}$ = number of cycles remaining for a flaw to grow from some initial size ($\sim K_{Ii}$) to the threshold size ($\sim K_{TH}$);

K_{Ic} = plane strain fracture toughness of the material;

K_{TH} = threshold stress intensity factor;

α = proof test factor;

σ_{proof} = proof test stress level;

$C^* = 1.21 \pi M_k^2 C$;

C = upper bound flaw-growth coefficient;

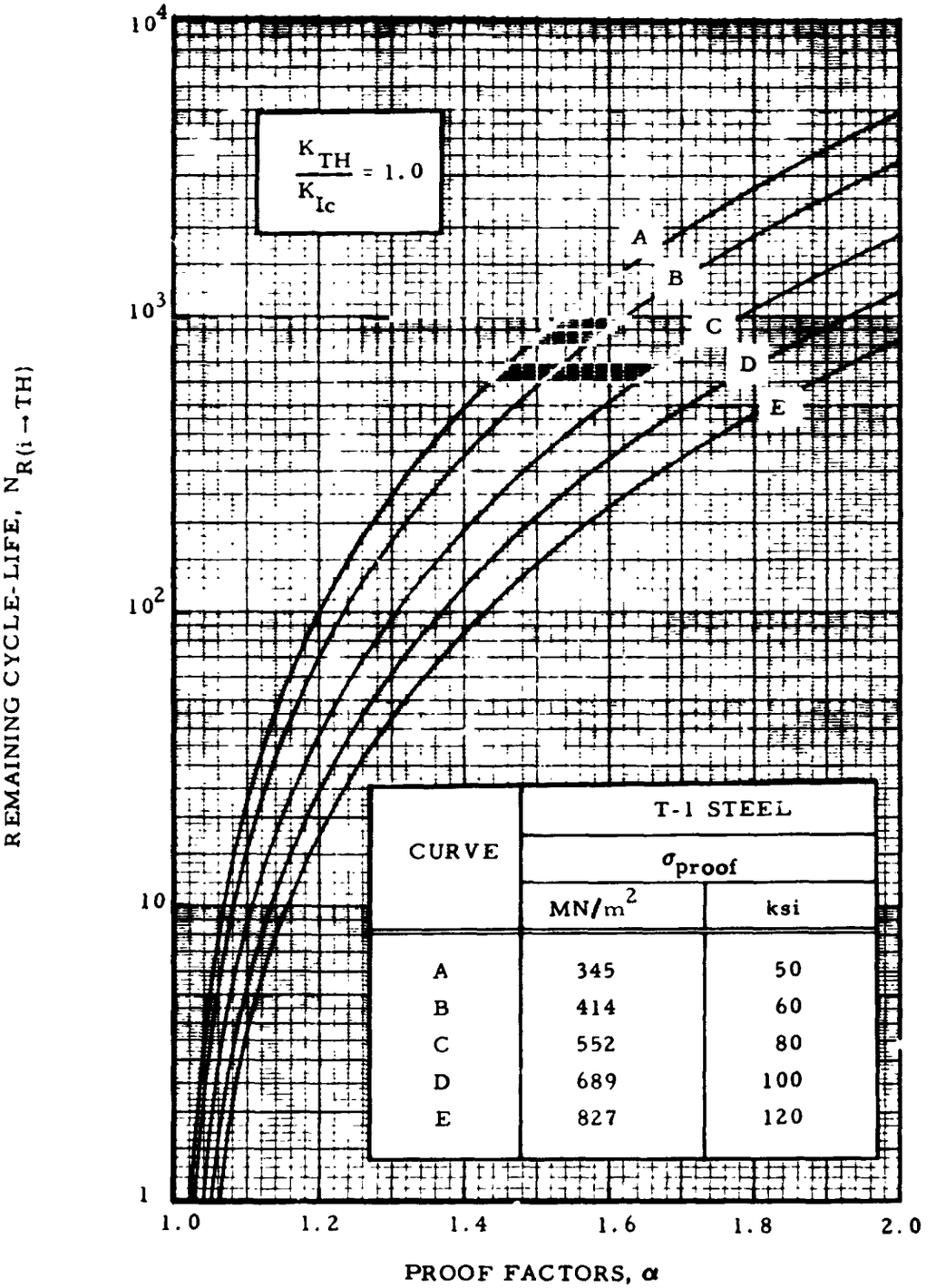
M_k = deep flaw magnification factor;

= 1.60 for this study.

In Figures D-4, D-5, and D-6, Equation (3) is displayed graphically for several values of σ_{proof} and K_{TH}/K_{Ic} for T-1 steel.

4. PROOF TEST PHILOSOPHY

Equation (3) can be used with the appropriate proof test requirements to establish the appropriate proof test factors. For single-use aerospace pressure vessels, the proof test requirements are well known. However, there are no established proof test requirements for reusable pressure vessels such as those intended for the Sortie mission and those designed for on-orbit maintenance and ground refurbishment. In the following paragraphs, several proof test philosophies are advanced for consideration along with the proof test requirement for expendable pressure vessels.



CURVE	T-1 STEEL σ_{proof}	
	MN/m^2	ksi
A	345	50
B	414	60
C	552	80
D	689	100
E	827	120

Figure D-4. Remaining Cycle Life for Given Initial Flaw Size

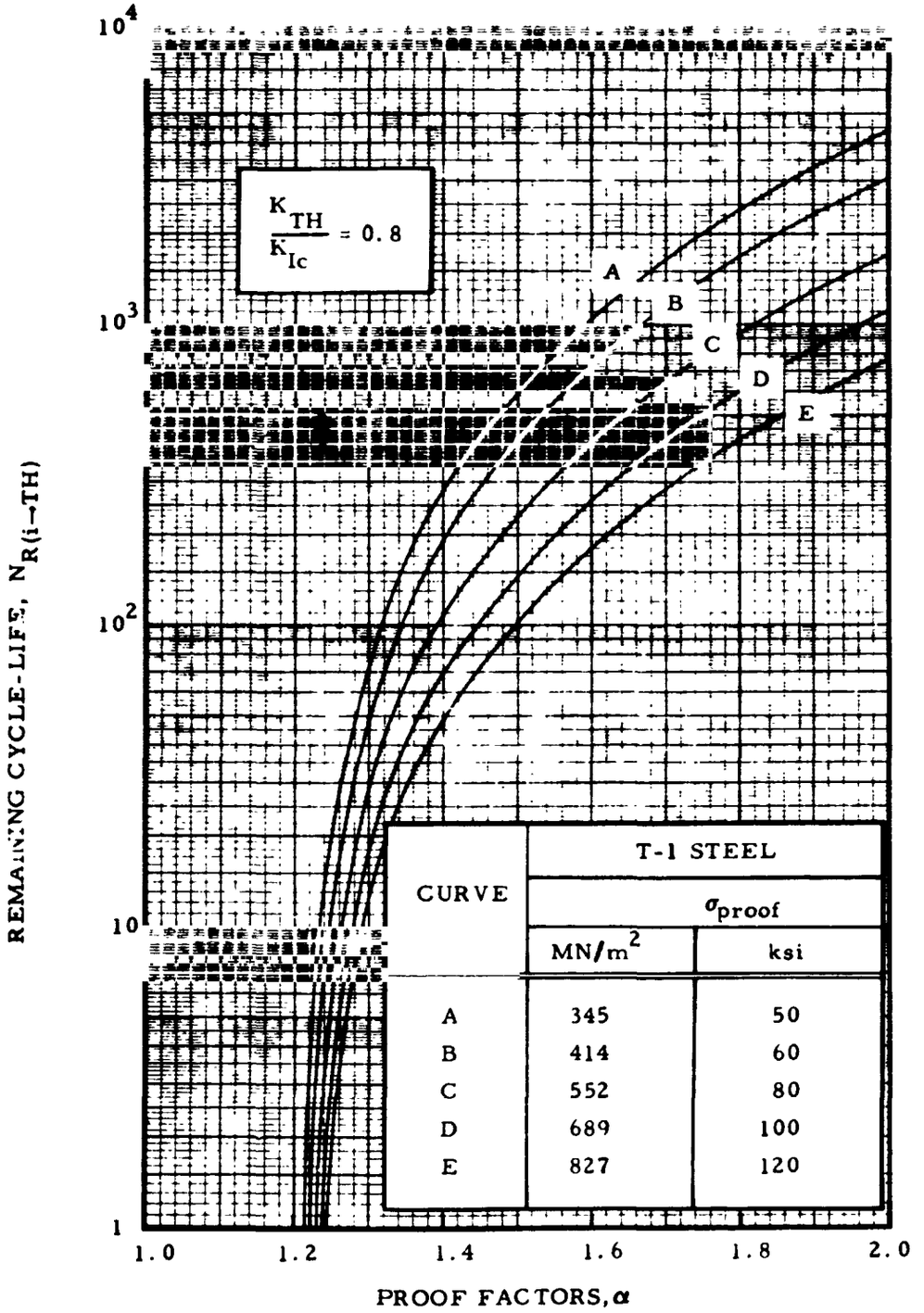


Figure D-5. Remaining Cycle Life for Given Initial Flaw Size

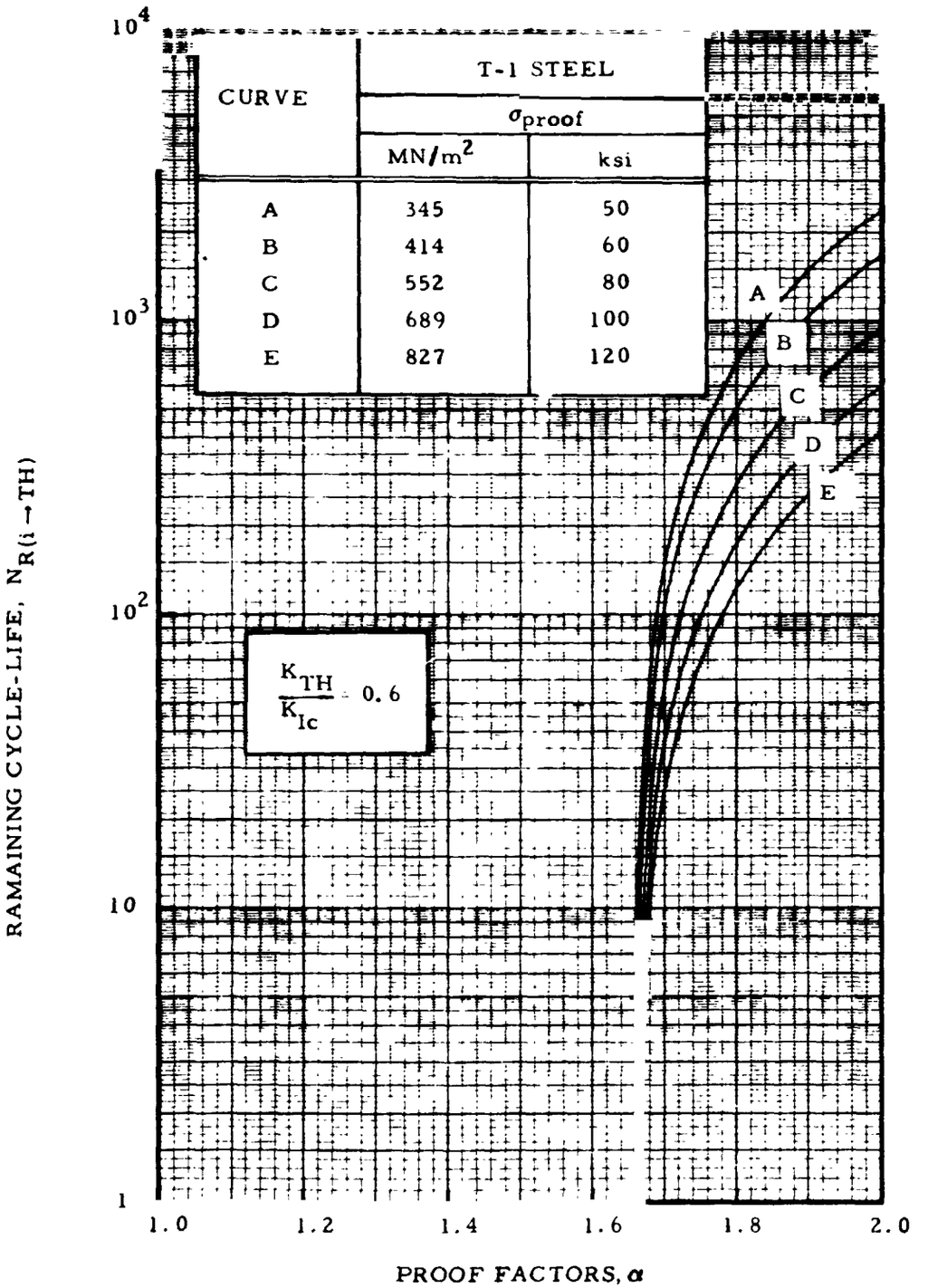


Figure D-6. Remaining Cylce Life for Given Initial Flaw Size

a. Pressure Vessels for Expendable Payloads

Consistent with established "safe-life" design philosophy, these pressure vessels shall be proof-pressure tested before use to demonstrate a safe-life of at least four pressure cycles to mean effective operating pressure (MEOP).

b. Pressure Vessels for On-Orbit Maintainable Payloads

Since the material performance and degradation (if any) in space environment for long periods of time are not known at this time, it is proposed that allowances be provided to account for this possibility. It is also hypothesized that the frequency of retrieval of these pressure vessels for maintenance is no more than once every two to three years. It follows, therefore, that these pressure vessels are readily demated from the payloads such that failure of the vessels during hydrostatic proof test will not damage the rest of the payload system. The traditional concept of safe life design to at least four times the specified service life is enforced.

The above considerations led to the formulation of the proof test philosophy that pressure vessels designed for on-orbit maintainable payloads shall be proof-pressure tested initially, before the vessels are put to service, to demonstrate a safe-life of 20 pressure cycles to MEOP. Thereafter, these pressure vessels shall be proof tested to demonstrate a safe-life of 16 pressure cycles to MEOP after the first use, 12 pressure cycles to MEOP after the second use, and 8 pressure cycles to MEOP after the third and subsequent uses.

c. Pressure Vessels for Ground-Refurbishable Payloads

Ground-refurbishable payloads involve retrieval of the entire payload for ground refurbishment. In this context, two distinct classes of pressure vessels must be considered: vessels which can be demated from the paylcads and vessels which are integral with the payloads.

Consider first dematable pressure vessels. Their proof test requirements should not differ substantially from those designed for on-orbit maintainable payloads. Therefore, for this study, these two types of pressure vessels are equated as identical.

On the other hand, vessels that are integral with the payloads should not be expected to undergo major refurbishment after each use. Indeed, any refurbishment should be limited to normal maintenance and resupply function. Clearly, catastrophic failures of integrally designed pressure vessels during proof tests are intolerable. However, the possibilities of failure during proof test always exist from errors in design, manufacturing, undetected flaws, etc. The recognition of such possibilities therefore requires that these "safe-life" designed pressure vessels also be "fail safe." In this context, fail-safe behavior of a pressure vessel is defined as a "leak-before-burst" non-catastrophic failure mode resulting from a localized, non-propagating rupture of the vessel wall. The appropriate proof test philosophy must therefore require that the pressure vessel be proof tested initially to demonstrate compliance with expected service life, followed by proof test after each use to verify a safe life of four pressure cycles to MEOP. This proof test philosophy is based on the argument that successful compliance with the initial proof test will ensure that the vessels will meet their service life requirements if no new flaws are introduced during operation, and the subsequent proof tests will ensure that no detrimental new flaws are intorduced during the previous use.

d. Pressure Vessels for Sortie Payloads

By definition, Sortie payloads return to their launch site at frequent intervals. Minimum maintenance and repair therefore important requisites for pressure vessels designed for this purpose. It is also surmised that these vessels are exposed to relatively benign environments and are housed protectively within the spacecraft. Under these circumstances, it appears reasonable to postulate a proof-test

philosophy requiring only an initial proof test before the vessels are put to service, to a proof pressure sufficient to demonstrate a safe life of four times the specified service life. Obviously, if the assumed conditions are violated, reproof testing may be necessary and is, in fact, recommended to assure operational success.

5. FACTORS OF SAFETY

The significance of the proof test on the structural design and structural performance of pressure vessels has previously been discussed in detail by Tiffany (Ref. D-8) and also by Au and Spear (Ref. D-3). Briefly, if a pressure vessel successfully sustains an elastic proof stress of

$$\sigma_{proof} = \alpha \sigma_{operating} \qquad (\alpha > 1)$$

where α is the proof-test factor, the maximum initial flaw size which can exist immediately after the test is $1/\alpha^2$ of the critical size at the operating pressure. The vessel will fail if the flaw grows from this initial value to the critical value. In addition, a successful proof test indicates that the maximum initial-to-critical stress intensity ratio K_{Ii}/K_{Ic} is equal to $1/\alpha$. Accordingly, a properly designed and successfully executed proof-pressure test is perhaps the most reliable nondestructive inspection technique available for insuring that there are no initial flaws in the vessel of sufficient size to cause failure under operating conditions.

The appropriate proof factors for the four types of pressure vessels can be quantified by suitably combining the aforementioned proof test philosophies with Equation (3) and limiting the stress level during proof test to the uniaxial yield strength of the material. These proof factors are tabulated in Tables D-2 to D-5, inclusively.

With a knowledge of the proof factors, the design factor of safety can be obtained simply from the equation

$$F.S. = \alpha \frac{F_{tu}}{F_{ty}} \qquad (4)$$

Table D-2. Proof Factors and Design Factors of Safety

PROOF TEST PHILOSOPHY: Proof test before use to demonstrate a safe life of four pressure cycles to MEOP.

APPLICATION: Pressure vessels for expendable payloads.

	ALUMINUM				TITANIUM				STEEL			
$\frac{K_{TH}}{K_{Ic}}$	2219-T851		6061-T651		5Al-2.5Sn		6Al-4V		T-1		HP9-4-0.25	
	α	F.S.	α	F.S.	α	F.S.	α	F.S.	α	F.S.	α	F.S.
1.0	1.22	1.33	1.10	1.28	1.17	1.26	1.12	1.25	1.10	1.20	1.18	1.23
0.8	1.33	1.45	1.26	1.47	1.29	1.38	1.27	1.42	1.26	1.38	1.31	1.36
0.6	1.69	1.84	1.67	1.95	1.67	1.79	1.66	1.85	1.67	1.83	1.68	1.75

Table D-3. Proof Factors and Design Factors of Safety

PROOF TEST PHILOSOPHY: Proof test initially, before the vessel is put to service, to a proof pressure sufficient to demonstrate a safe life of 20 pressure cycles to MEOP. Thereafter, the pressure vessel shall be proof tested to demonstrate a safe life of:

- o 16 pressure cycles to MEOP after the first use
- o 12 pressure cycles to MEOP after the second use
- o 3 pressure cycles to MEOP after the third and subsequent uses.

APPLICATION: Pressure vessels for on-orbit maintainable payloads and ground-refurbishable payloads in which the pressure vessels can be demated from the payloads during proof testing.

MATERIAL	$\frac{K_{TH}}{K_{Ic}}$	PROOF FACTORS				F. S.
		α_1	α_2	α_3	$\alpha_{4...}$	
2219-T851	1	1.45	1.40	1.36	1.30	1.58
6061-T651		1.20	1.18	1.16	1.13	1.40
5Al-2.5Sn		1.32	1.30	1.26	1.22	1.42
6Al-4V		1.26	1.24	1.21	1.17	1.40
T-1		1.21	1.19	1.16	1.13	1.32
HP9Ni-4Co-0.25C		1.36	1.33	1.30	1.25	1.41
2219-T851	0.8	1.52	1.48	1.44	1.39	1.65
6061-T651		1.31	1.30	1.28	1.27	1.53
5Al-2.5Sn		1.40	1.38	1.35	1.32	1.50
6Al-4V		1.35	1.34	1.32	1.30	1.50
T-1		1.33	1.31	1.29	1.27	1.45
HP9Ni-4Co-0.25C		1.45	1.42	1.39	1.35	1.51
2219-T851	0.6	1.80	1.77	1.75	1.72	1.96
6061-T651		1.69	1.69	1.68	1.67	1.97
5Al-2.5Sn		1.73	1.71	1.70	1.68	1.86
6Al-4V		1.70	1.70	1.69	1.67	1.90
T-1		1.69	1.68	1.67	1.66	1.85
HP9Ni-4Co-0.25C		1.75	1.73	1.72	1.70	1.82

Table D-4. Proof Factors and Design Factors of Sa

PROOF TEST PHILOSOPHY: Proof test initially to demonstrate compliance with
test after each use to verify a safe life of four pre

APPLICATION: Pressure vessels for ground-refurbishable payloads in which the
with the spacecraft during proof testing.

N_{uses}	$\frac{K_{TH}}{K_{Ic}}$	ALUMINUM						TITANIUM					
		2219-T851			6061-T651			5Al-2.5Sn			6Al-4V		
		α_i	α_s	F.S.	α_i	α_s	F.S.	α_i	α_s	F.S.	α_i	α_s	
5	1.0	1.56	1.17	1.70	1.24	1.08	1.45	1.38	1.12	1.48	1.33	1.11	1
	0.8	1.83	1.30	2.00	1.63	1.26	2.08	1.72	1.28	1.84	1.68	1.27	1.
	0.6	2.91	1.67	3.16	2.79	1.67	3.26	2.84	1.67	3.05	2.86	1.68	3
10	1.0	1.68	1.16	1.83	1.30	1.08	1.51	1.48	1.12	1.58	1.40	1.10	1
	0.8	1.95	1.29	2.12	1.66	1.26	2.12	1.79	1.28	1.92	1.74	1.27	1
	0.6	3.00	1.67	3.27	2.82	1.67	3.29	2.88	1.67	3.09	2.89	1.68	3
15	1.0	1.76	1.15	1.92	1.33	1.07	1.55	1.53	1.11	1.64	1.45	1.10	1
	0.8	2.04	1.28	2.22	1.70	1.26	2.17	1.84	1.27	1.98	1.78	1.27	1
	0.6	3.08	1.67	3.35	2.84	.67	3.31	2.92	1.67	3.13	2.91	1.68	3
20	1.0	1.84	1.15	2.00	1.36	1.07	1.59	1.58	1.10	1.69	1.48	1.09	1
	0.8	2.10	1.27	2.28	1.73	1.26	2.21	1.91	1.27	2.04	1.82	1.26	2
	0.6	3.16	1.67	3.44	2.86	1.67	3.34	2.96	1.66	3.17	2.94	1.68	3
25	1.0	1.88	1.14	2.05	1.39	1.07	1.62	1.62	1.10	1.74	1.52	1.09	1
	0.8	2.16	1.27	2.35	1.74	1.26	2.23	1.94	1.27	2.08	1.87	1.26	2
	0.6	3.21	1.67	3.50	2.87	1.67	3.35	3.01	1.66	3.23	2.96	1.68	3

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- Proof Factors and Design Factors of Safety

initially to demonstrate compliance with expected service life followed by proof each use to verify a safe life of four pressure cycles to MEOP.

round-refurbishable payloads in which the pressure vessels are integral ring proof testing.

	TITANIUM						STEEL					
	5Al-2.5Sn			6Al-4V			T-1			HP9-4-0.25		
S.	α_i	α_s	F. S.	α_i	α_s	F. S.	α_i	α_s	F. S.	α_i	α_s	F. S.
5	1.38	1.12	1.48	1.33	1.11	1.48	1.27	1.09	1.38	1.45	1.14	1.51
3	1.72	1.28	1.84	1.68	1 27	1.87	1.60	1.24	1.75	1.75	1.28	1.82
.6	2.84	1.67	3.05	2.86	1.68	3.19	2.79	1.66	3.05	2.66	1.67	2.97
1	1.48	1.12	1.58	1.40	1.10	1.56	1.32	1.09	1.44	1.55	1.13	1.61
2	1.79	1.28	1.92	1.74	1.27	1.94	1.65	1.24	1.80	1.83	1.28	1.90
.9	2.88	1.67	3.09	2.89	1.68	3.22	2.82	1.66	3.08	2.92	1.67	3.04
55	1.53	1.11	1.64	1.45	1.10	1.62	1.35	1.08	1.48	1.61	1.12	1.68
.7	1.84	1.27	1.98	1.78	1.27	1.98	1.69	1.24	1.84	1.89	1.27	1.97
.1	2.92	1.67	3.13	2.91	1.68	3.24	2.84	1.66	3.11	2.97	1.67	3.09
9	1.58	1.10	1.69	1.48	1.09	1.66	1.37	1.07	1.50	1.67	1.12	1.74
.1	1.91	1.27	2.04	1.82	1.26	2.03	1.71	1.24	1.87	1.95	1.27	2.03
4	2.96	1.66	3.17	2.94	1.68	3.28	2.86	1.66	3.13	3.03	1.67	3.15
2	1.62	1.10	1.74	1.52	1.09	1.69	1.40	1.07	1.53	1.71	1.12	1.78
.3	1.94	1.27	2.08	1.87	1.26	2.08	1.74	1.24	1.90	2.01	1.27	2.09
5	3.01	1.66	3.23	2.96	1.68	3.30	2.87	1.66	3.14	3.07	1.67	3.19

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Table D-5. Proof Factors and Design Factors of Safety

PROOF TEST PHILOSOPHY: Proof test only once, before the vessel is put to service, to a proof pressure sufficient to demonstrate a safe life of four times the specified service life.

APPLICATION: Pressure vessels for Sortie payloads.

N_{uses}	$\frac{K_{TH}}{K_{Ic}}$	ALUMINUM				TITANIUM				STEEL			
		2219-T851		6061-T651		5Al-2.5Sn		6Al-4V		T-1		HP9-4-0.25	
		α	F.S.	α	F.S.	α	F.S.	α	F.S.	α	F.S.	α	F.S.
1	1.0	1.22	1.33	1.10	1.28	1.17	1.26	1.12	1.25	1.10	1.20	1.18	1.23
	0.8	1.33	1.45	1.26	1.47	1.29	1.38	1.27	1.42	1.26	1.38	1.31	1.36
	0.6	1.69	1.84	1.67	1.95	1.67	1.79	1.66	1.85	1.67	1.83	1.68	1.75
10	1.0	1.60	1.74	1.27	1.48	1.43	1.53	1.36	1.52	1.29	1.41	1.49	1.55
	0.8	1.65	1.79	1.37	1.60	1.50	1.61	1.43	1.59	1.38	1.51	1.55	1.61
	0.6	1.89	2.06	1.71	1.99	1.78	1.91	1.75	1.95	1.70	1.86	1.82	1.89
25	1.0	1.86	2.02	1.40	1.63	1.63	1.75	1.52	1.69	1.43	1.56	1.72	1.79
	0.8	1.90	2.07	1.47	1.71	1.69	1.81	1.58	1.76	1.49	1.63	1.76	1.83
	0.6	2.08	2.26	1.75	2.04	1.90	2.04	1.85	2.06	1.78	1.95	1.98	2.06
50	1.0	2.12	2.31	1.53	1.78	1.83	1.96	1.70	1.90	1.57	1.72	1.94	2.02
	0.8	2.17	2.36	1.58	1.84	1.87	2.01	1.74	1.94	1.62	1.77	1.97	2.05
	0.6	2.31	2.51	1.84	2.15	2.05	2.20	1.97	2.20	1.86	2.03	2.15	2.24
100	1.0	2.45	2.66	1.70	1.98	2.07	2.22	1.91	2.13	1.75	1.91	2.20	2.29
	0.8	2.49	2.71	1.74	2.03	2.11	2.26	1.95	2.17	1.79	1.96	2.24	2.33
	0.6	2.60	2.83	1.96	2.29	2.26	2.42	2.13	2.37	1.98	2.16	2.38	2.47

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for all pressure vessel types previously discussed except those pressure vessels which are integral with the payloads. For this configuration, the appropriate design factor of safety is the larger of

$$F.S. = \alpha_i \frac{F_{tu}}{F_{ty}} \tag{5}$$

$$\text{or} \qquad F.S. = \alpha_i^2 \left(\frac{1.21\pi M_k^2}{Q} \right) \left(\frac{F_{tu}}{K_{Ic}^2} \right) \left(Cp_{meop} R \right) \tag{6}$$

Equation (6) defines the minimum factor of safety required for fail safe or leak-before-burst behavior at the proof pressure level. Its derivation is detailed under "Fail-Safe Design" below.

Values of the appropriate design factor of safety for safe-life design of the four types of pressure vessels are summarized in Tables D-2 through D-5, inclusively. They are also graphically displayed in Figures D-7 to D-10, inclusively. From Figure D-7, it is seen that the conventional, although arbitrarily chosen, values of 1.25-1.40 for expendable pressure vessels are quite appropriate for this application when $K_{TH}/K_{Ic} \rightarrow 1.0$ for single use.

For fail-safe design of pressure vessels which are integral with the payloads during proof testing, the design factors of safety are shown graphically in Figures D-11 to D-15, inclusively, for the materials under investigation and for $K_{TH}/K_{IC} = 0.8$. Similar design curves can be generated for other values of K_{TH}/K_{Ic} by combining the results of Table D-4 with Equation (6).

6. FAIL-SAFE DESIGN: LEAK-BEFORE-BURST CRITERION

In Reference D-9, an equation defining the minimum factor of safety required in fail-safe or leak-before-burst behavior at the operating pressure level was derived by Olsen and Speare. This equation is

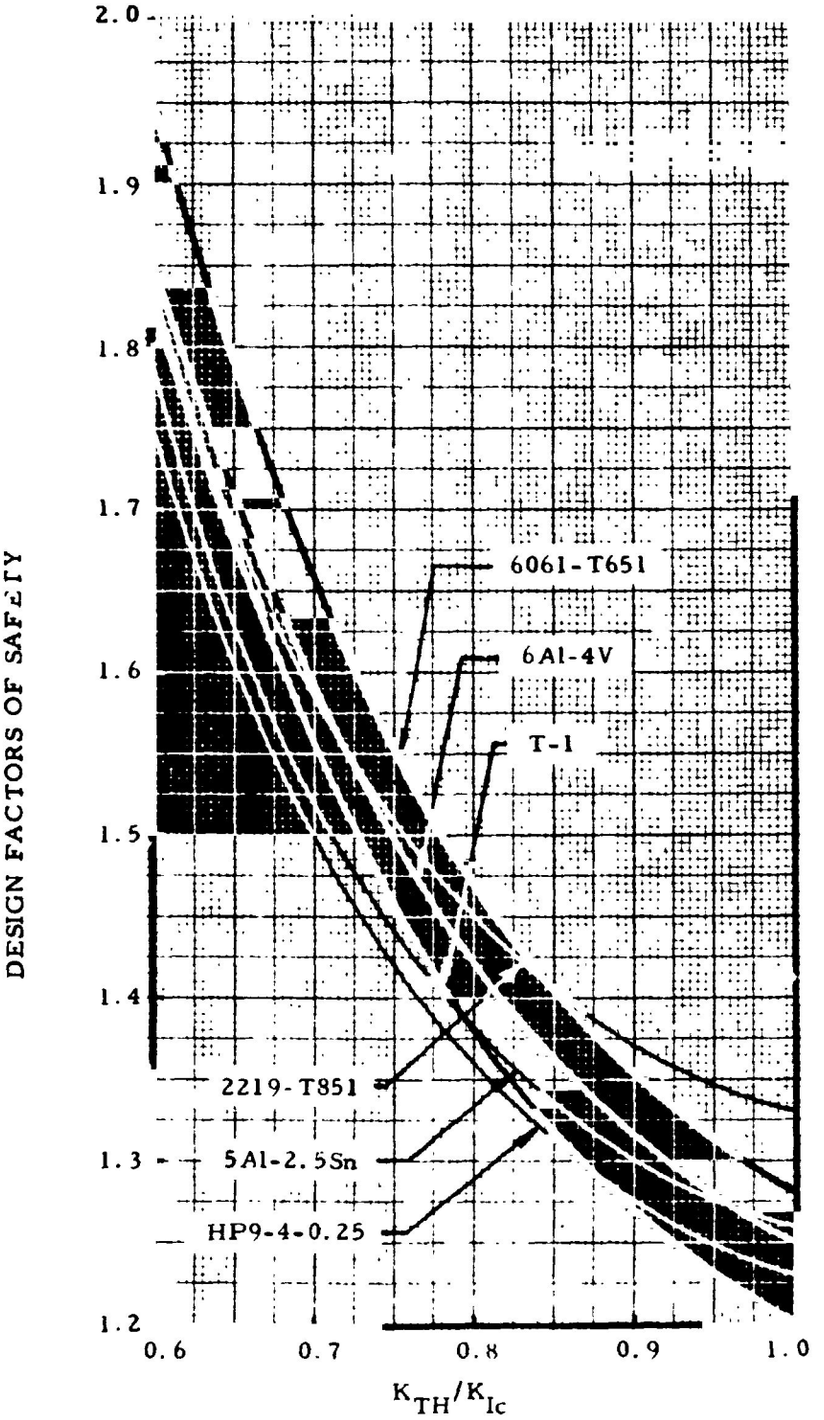


Figure D-7. Design Factors of Safety for Safe-Life Design of Expendable Pressure Vessels

• PRESSURE VESSELS ARE NOT DEMATED FROM SPA

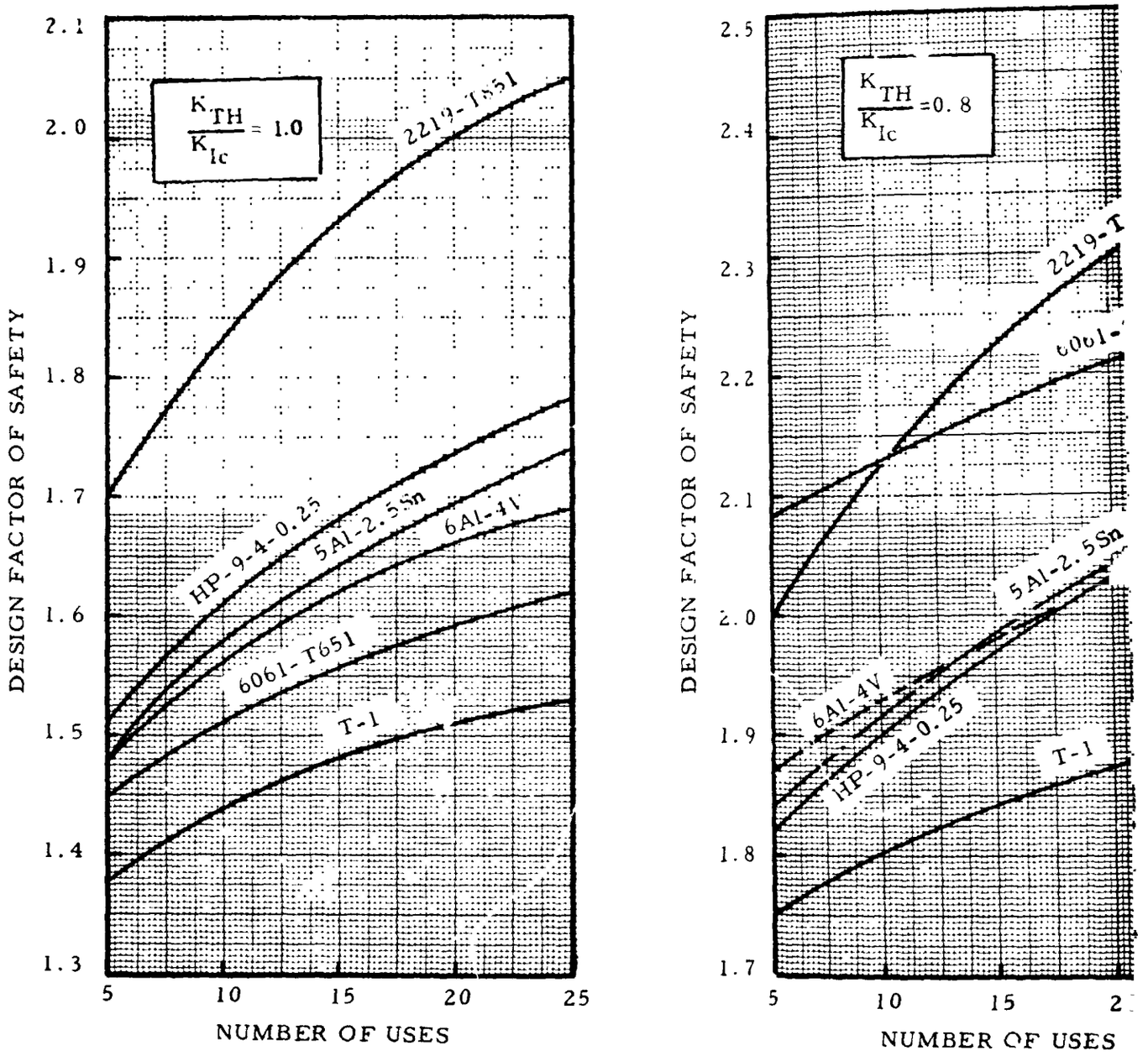
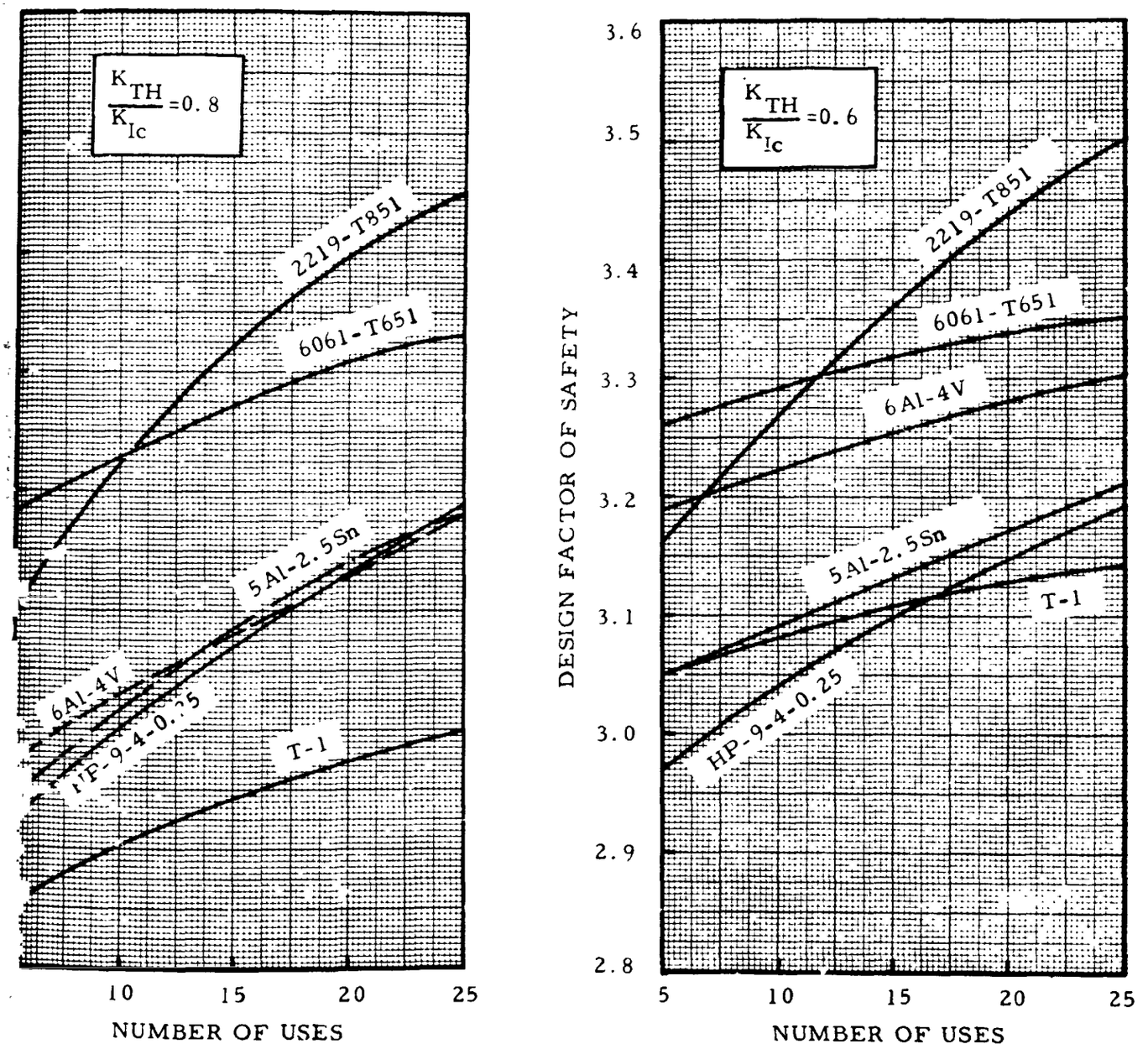


Figure D-9. Design Factors of Safety for Sa...-Life Design of Ground

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Safe-Life Design of Ground-Refurbishable Pressure Vessels

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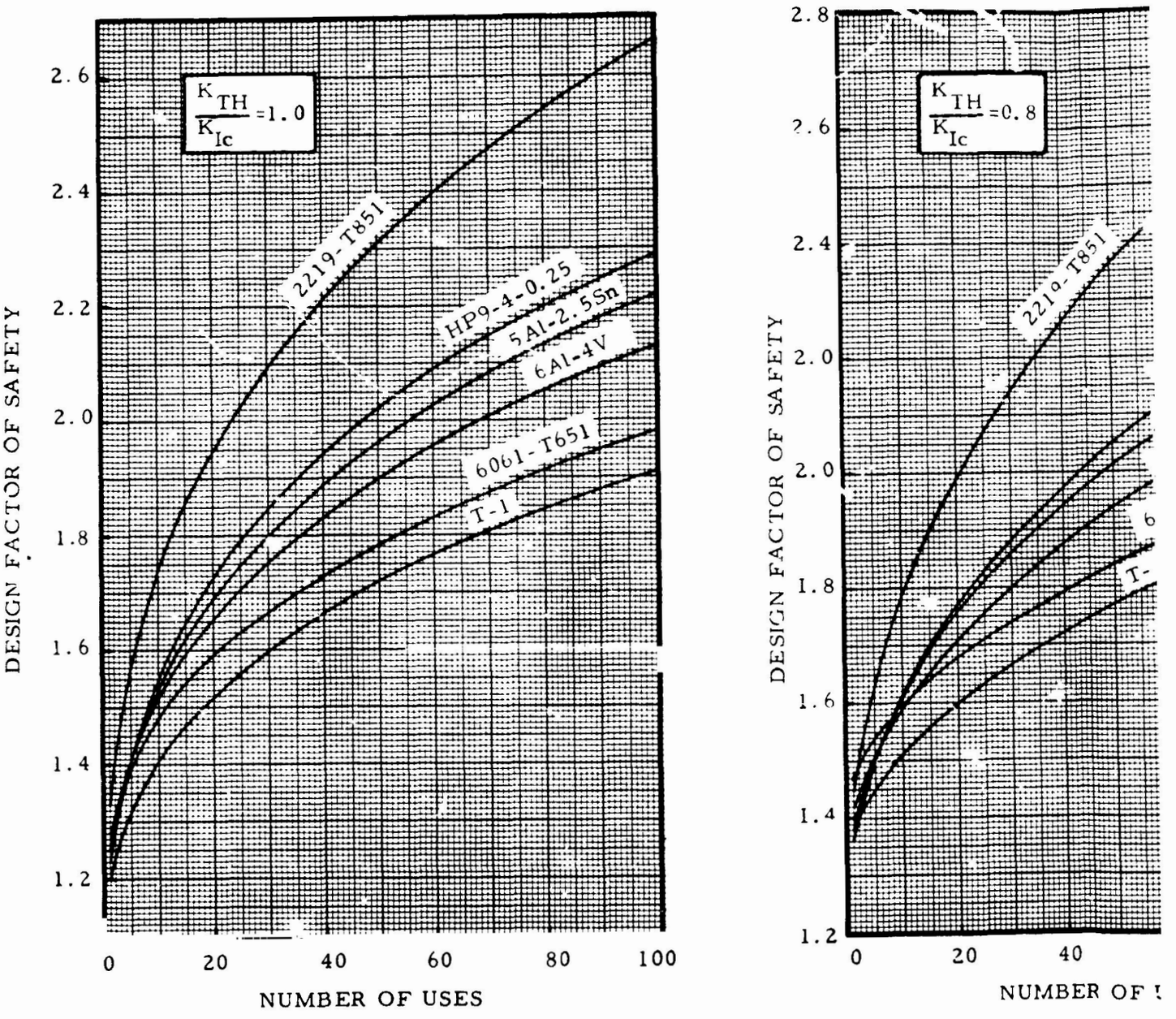
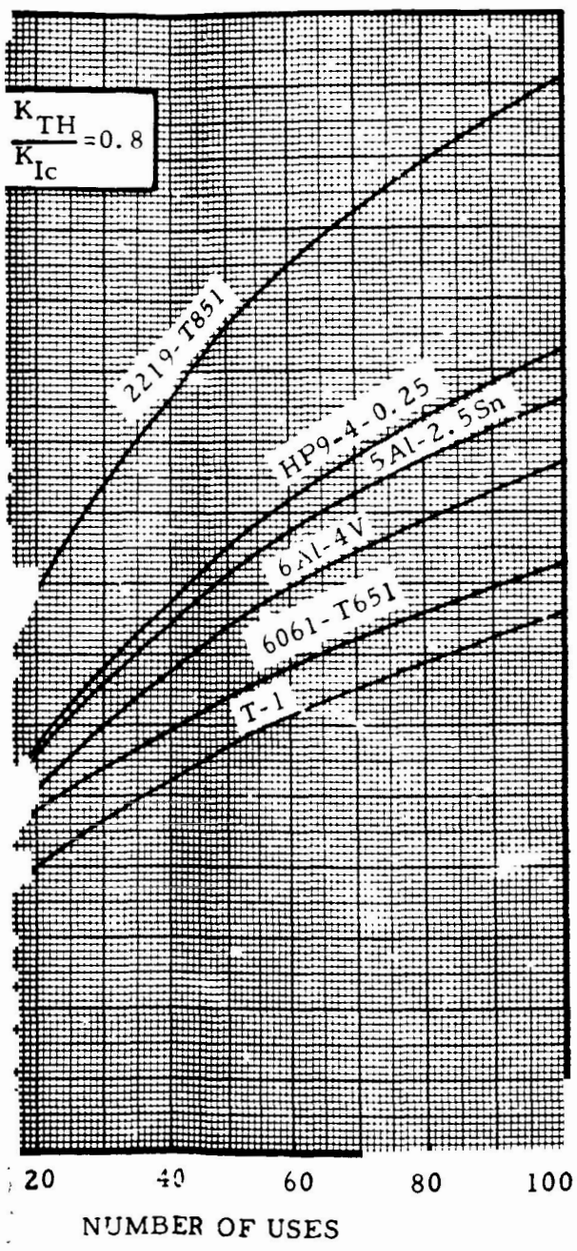
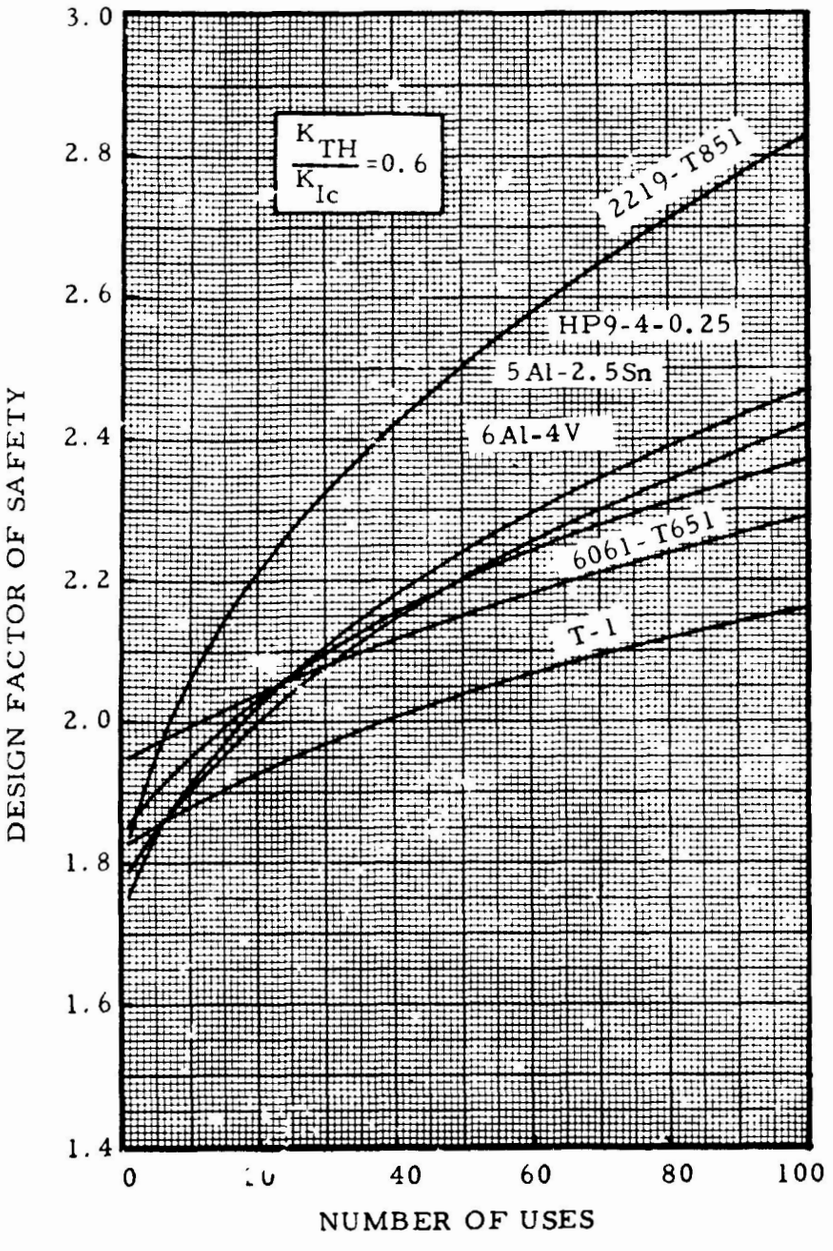


Figure D-10. Design Factors of Safety for Safe-Life I

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'ety for Safe-Life Design of Sortie Mission Pressure Vessels

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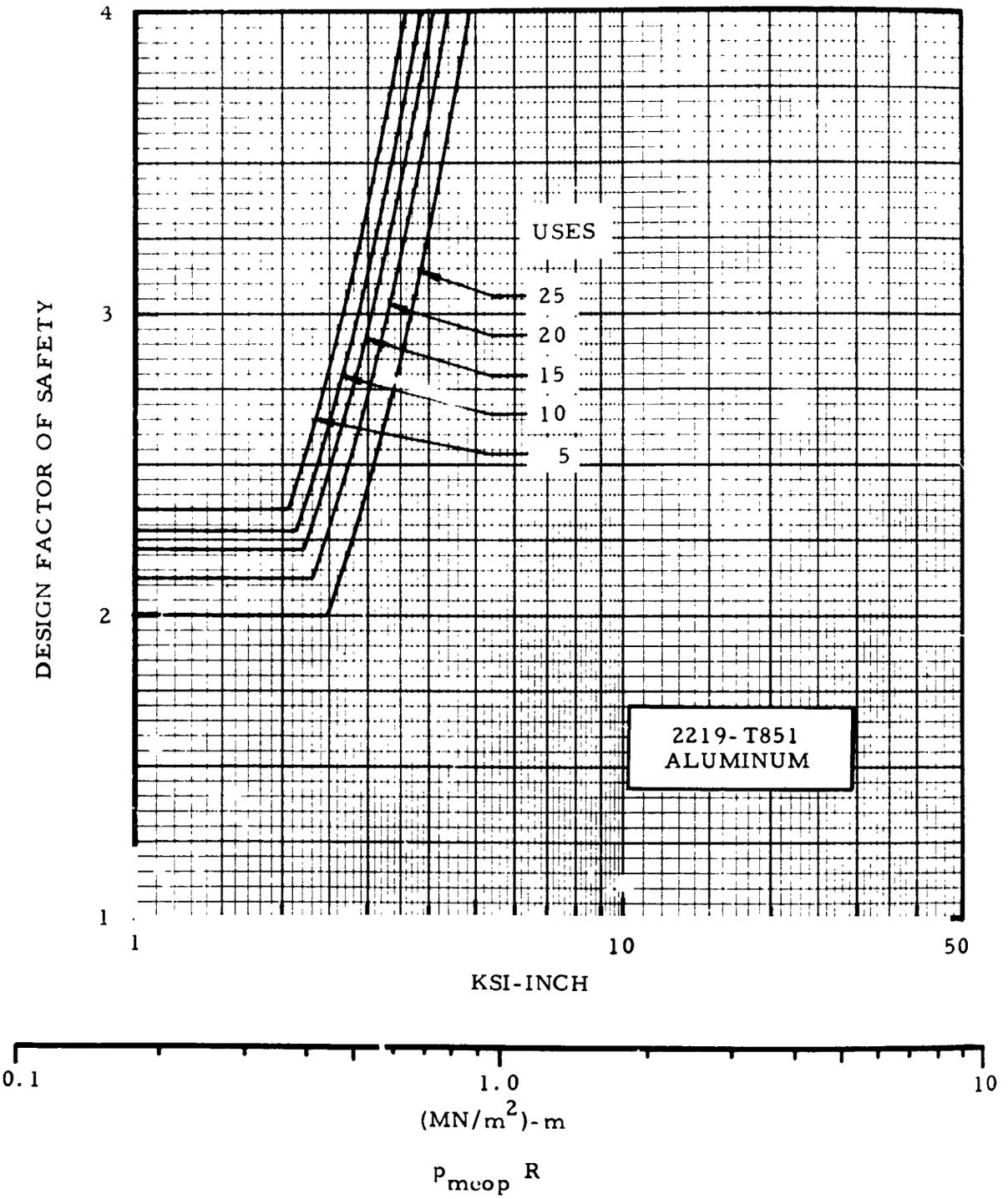
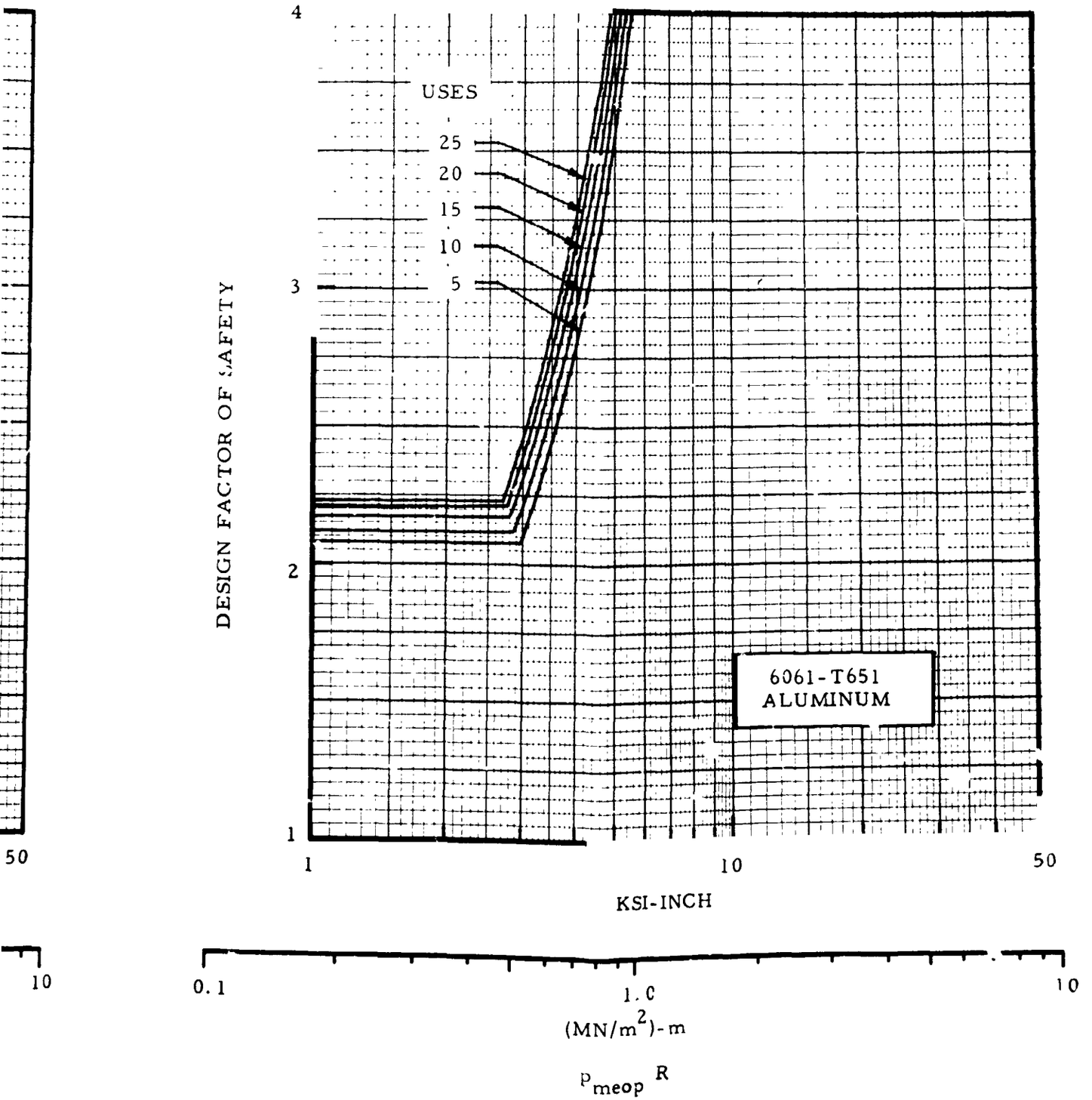


Figure D-11. Design Factors of Safety for Fail-Safe
Which are Integral with Spacecraft dur'

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r Fail-Safe Design of Ground-Refurbishable Pressure Vessels

acecraft during Proof Testing (K_{TH}/K_{Ic} = 0.8)

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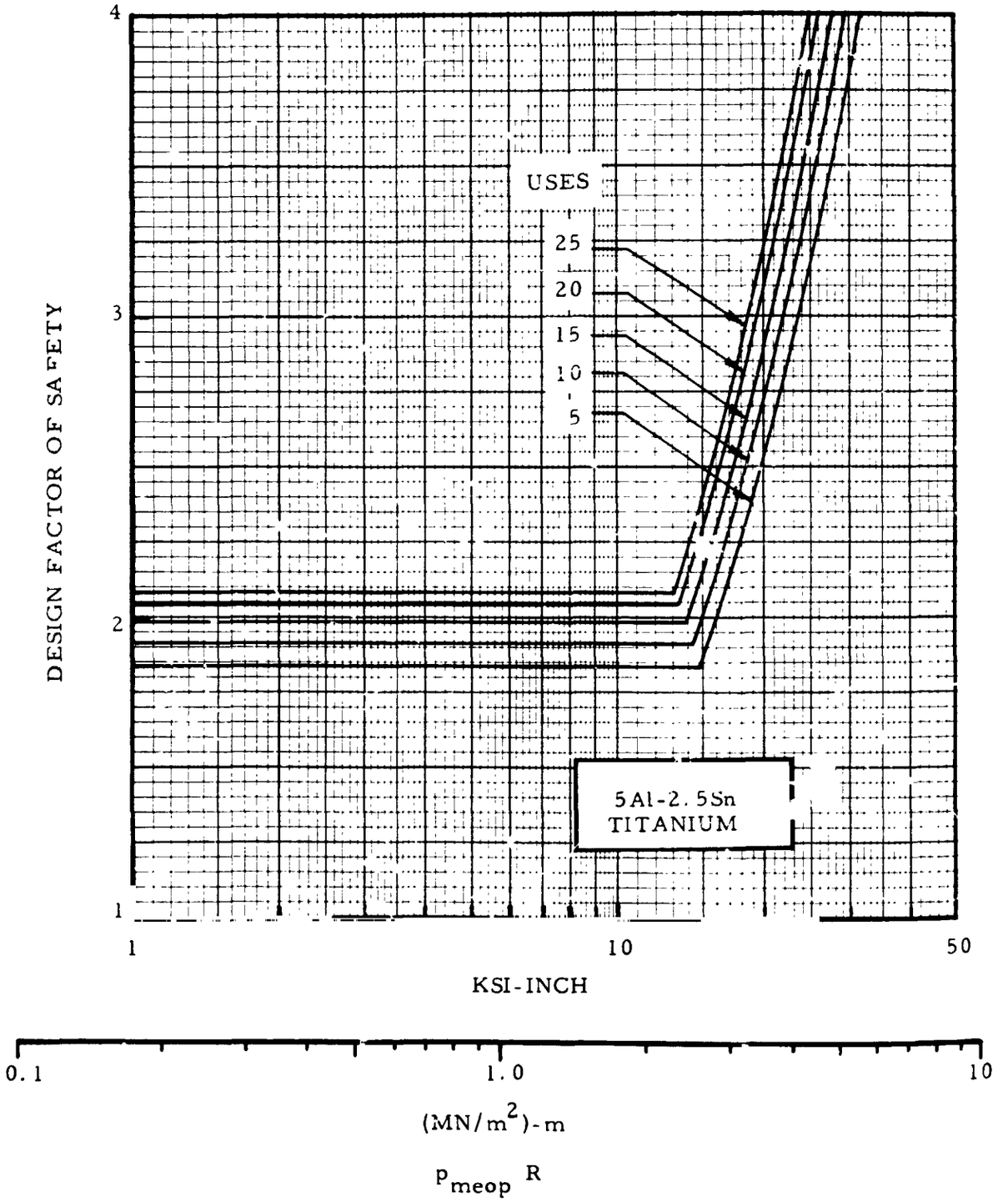
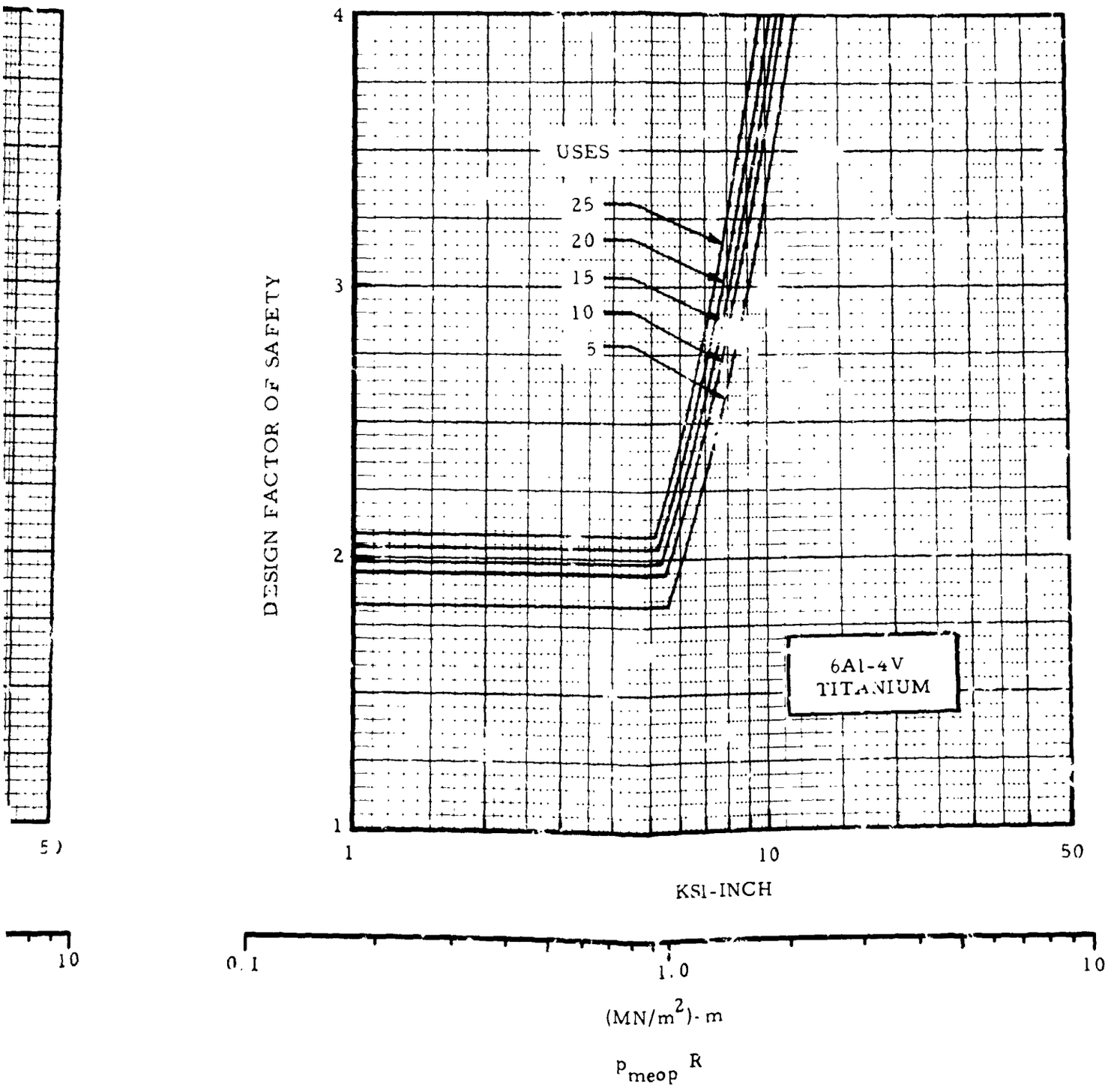


Figure D-12. Design Factors of Safety for Fail-Safe De
Which are Integral with Spacecraft during

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:y for Fail-Safe D. sign of Ground-Refurbishable Pres.. e Vessels
ı Spacecraft during Proof Testing (K_{TH}/K_{Ic} = 0.8)

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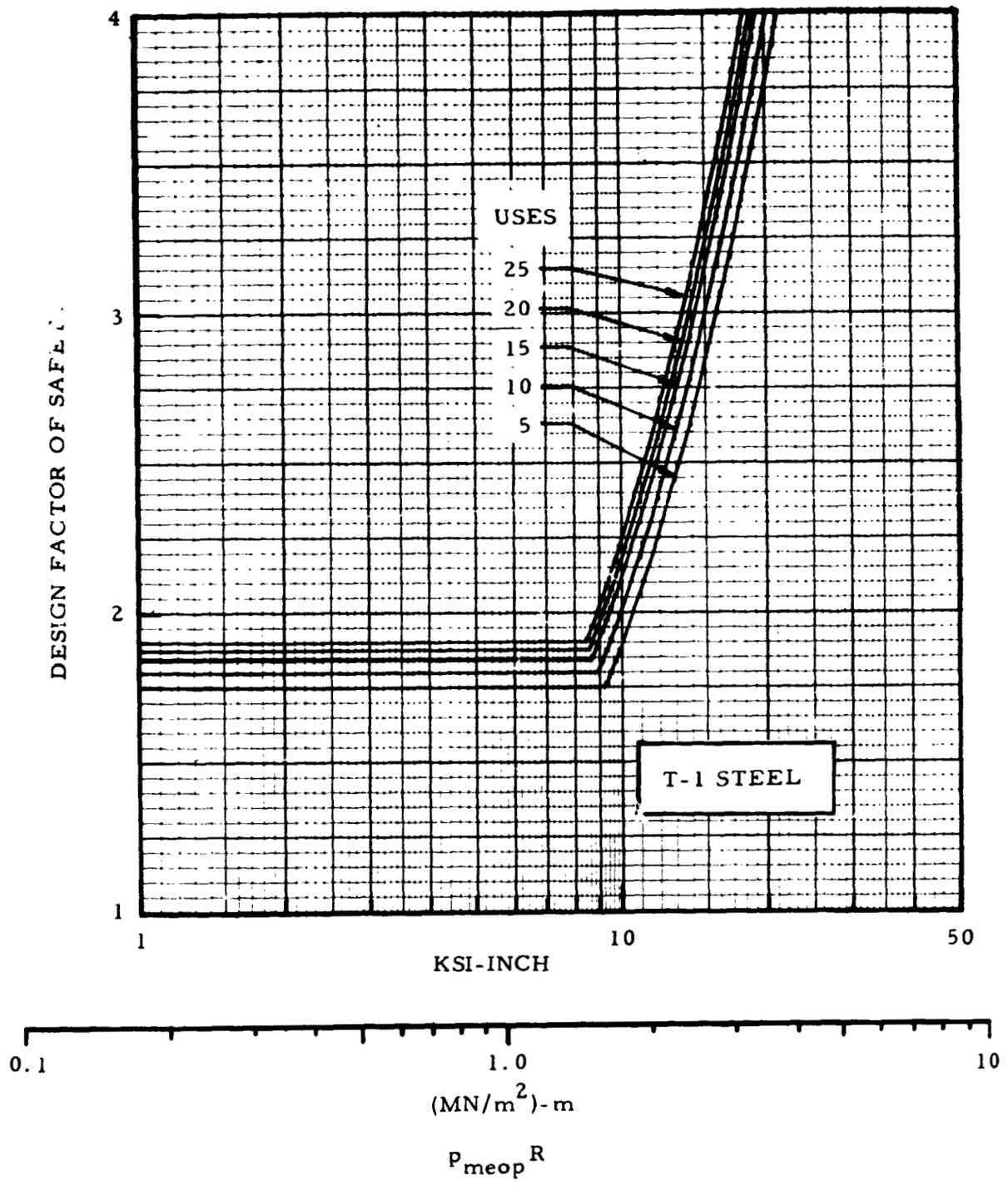
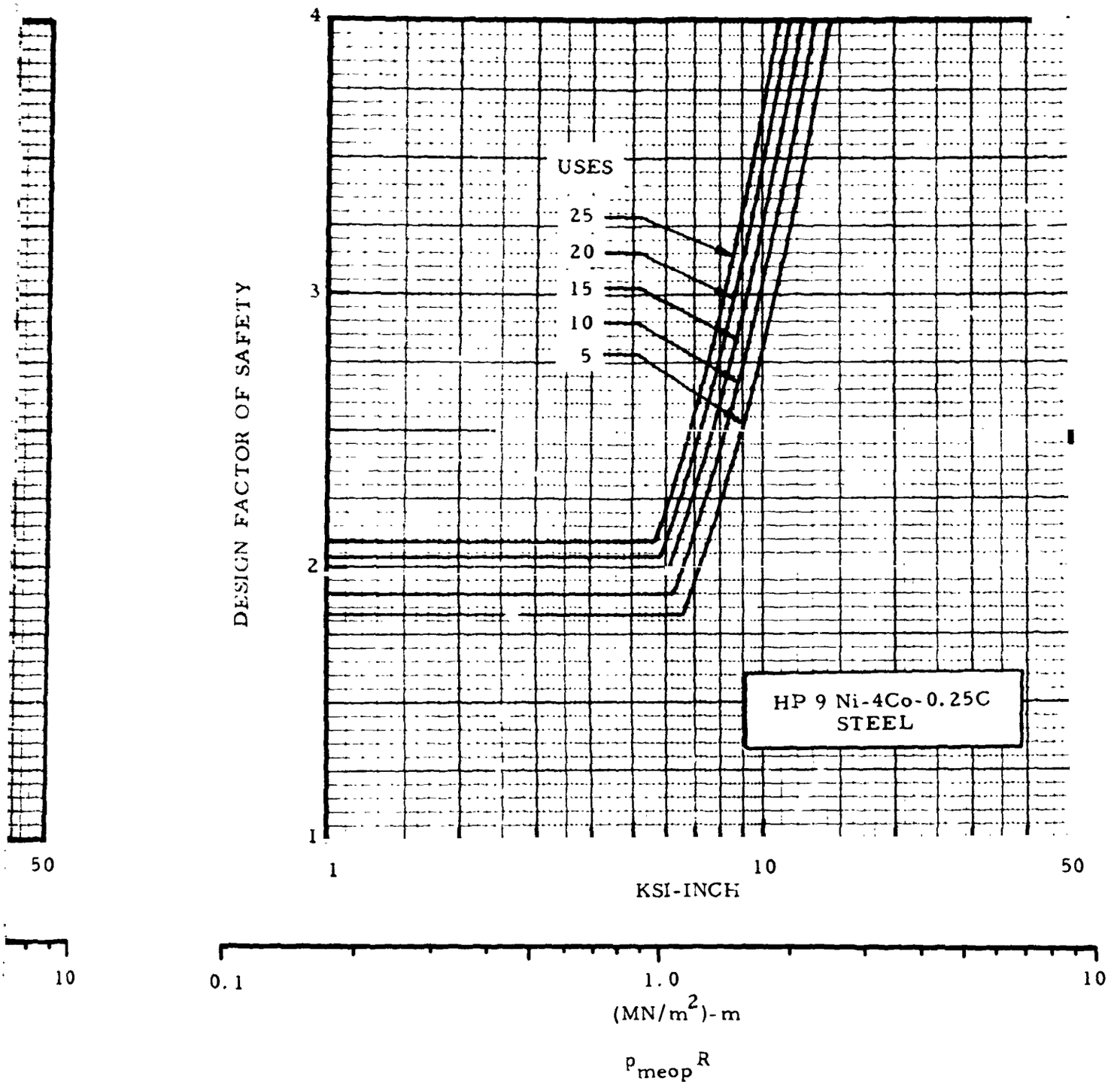


Figure D-13. Design Factors of Safety for Fail-Safe
Which are Integral with Spacecraft dur

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ety for Fail-Safe Desig.n of Ground-Refurbishable Pressure Vessels
.n Spacecraft during Proof Testing (K_{TH}/K_{Ic} = 0.8)

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$$(F.S.)_{FS_{min}} = \left[\left(\frac{1.21\pi M_k^2}{Q}\right)\left(\frac{F_{tu}}{K_{Ic}^2}\right)\left(Cp_{cap}R\right)\right]^{1/2} \qquad (7)$$

where M_k = stress intensity magnification factor, and is a function of the flaw depth-to-wall thickness ratio, a/t;

Q = flaw shape parameter, dimensionless;

F_{tu} = ultimate tensile strength of the material;

K_{Ic} = plane strain fracture toughness;

C = maximal value of stress coefficients in the pressure vessel;

= 1/2 for spherical pressure vessels;

p_{cap} = pressure-carrying capability of the vessel;

R = characteristic vessel radius.

Corresponding to Equation (7), the upper limit of pressure which will result in a fail-safe failure mode is

$$p_{FS_{max}} = \left[\left(\frac{Q}{1.21\pi M_k^2}\right)\left(\frac{K_{Ic}^2}{F_{tu}}\right)\left(\frac{p_{cap}}{CR}\right)\right]^{1/2} \qquad (8)$$

Clearly, $$p_{FS_{max}} \times (F.S.)_{FS_{min}} = p_{cap}$$

Defining the factor of safety as the ratio of the pressure that would cause failure of a vessel to the pressure that acts on it in service, it is seen that

$$F.S. = \frac{p_{cap}}{p_{meop}}$$

Accordingly, $(F.S.)\,(p_{meop}) = (F.S.)_{FS_{min}} \times p_{FS_{max}}$

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and $$F.S. = \left(\frac{p_{FS_{max}}}{p_{meop}}\right) (F.S.)_{FS_{min}} \quad (9)$$

As previously postulated, ground-refurbishable pressure vessels which are integral with the payload must not fail during proof testing. In other words, the maximum pressure experienced during proof test must not exceed the proper limits of that pressure which will result in a fail-safe failure mode. Thus, if α is the proof-test factor, the following relationship must hold to fulfill our objective:

$$p_{proof} = \alpha p_{meop} \leq p_{FS_{max}} \quad (\alpha > 1) \quad (10)$$

Equations (9) and (10) can be combined to give

$$F.S. = \alpha (F.S.)_{FS_{min}} \quad (11)$$

Accordingly, $$p_{cap} = \alpha p_{meop} (F.S.)_{FS_{min}} \quad (12)$$

When Equations (7) and (12) are substituted into Equation (11), the appropriate expression for the required factor of safety to insure fail-safe operation at the proof pressure level is obtained. Indeed,

$$\overline{F.S.} = \alpha^2 \left(\frac{1.21\pi M_k^2}{Q}\right)\left(\frac{F_{tu}}{K_{Ic}^2}\right)\left(Cp_{meop}R\right) \quad (13)$$

When pressure vessels are designed in accordance with Equation (13), the critical flaw depth at the proof pressure level should equal the thickness of the vessel wall. To prove the validity of this statement, we note the following relationships:

$$a_{cr} @ p_{proof} = \frac{Q}{1.21\pi M_k^2}\left(\frac{K_{Ic}}{\sigma_{proof}}\right)^2 \quad (14)$$

and

$$t = \frac{P_{cap} R}{C F_{tu}}$$

$$= \frac{F.S. \times p_{meop} R}{C F_{tu}}$$

$$= \left(\frac{1.21 \pi M_k^2}{Q} \right) \left(\frac{\alpha p_{meop} R}{C K_{Ic}} \right)^2$$

for a pressure vessel in the form of a surface of revolution.

But,

$$p_{proof} = \alpha p_{meop}$$

Hence,

$$t = \frac{1.21 \pi M_k^2}{Q} \left(\frac{p_{proof} R}{C K_{Ic}} \right)^2 \qquad (15)$$

From the familiar hoop stress equation $\sigma_{proof} = \frac{p_{proof} R}{Ct}$ for general pressure vessel shapes, we obtain

$$p_{proof} R = C t \sigma_{proof} \qquad (16)$$

When Equations (15) and (16) are combined and the results simplified, we find

$$t = \frac{Q}{1.21 \pi M_k^2} \left(\frac{K_{Ic}}{\sigma_{proof}} \right)^2$$

$$= a_{cr} \text{ @ } p_{proof} \quad \text{(Q.E.D.)}$$

The criteria for the usage of Equation (13) are that

- If F.S. $\geq \overline{F.S.}$, the potential failure mode is LEAK at the proof pressure level.
- If F.S. $< \overline{F.S.}$, the potential failure mode is BURST at the proof pressure level.

C. GUIDELINE APPLICATION

In the following paragraphs, the design guidelines previously developed will be applied to two example problems showing the effect on pressure vessel design of operating pressure and single versus multiple tank arrangements.

1. EXAMPLE 1: EFFECT OF OPERATING PRESSURE

The design requirements are as follows:

- Ground-refurbishable pressure vessels which are integral with the spacecraft during proof testing
- Uses: 15
- Operating pressure, p_{meop}: a. $0.689\ mn/m^2$ (100 psi)
 b. $6.895\ mn/m^2$ (1000 psi)
- Configuration: Spherical pressure vessel with R = 45.7 cm (18 in.)
- Material: T-1 Steel
- Environmental Effect: $K_{TH}/K_{Ic} = 0.8$

The design characteristics are summarized in Table D-6. It is seen that for Vessel A, satisfaction of the safe-life requirement automatically fulfills the fail-safe criterion, whereas, for Vessel B, the reverse situation exists. Furthermore, both vessels exhibit the leak-before-burst potential failure mode at the initial proof pressures. The critical flaw depths for the subsequent proof and operating pressure conditions are substantially larger than the wall thickness, insuring fail-safe failure modes for these conditions should undetected flaws grow to the critical size under abnormal circumstances.

2. EXAMPLE 2: SINGLE TANK VS. MULTIPLE TANKS

The design requirements are as follows:

- Ground-refurbishable pressure vessels which are integral with the spacecraft during proof testing

Table D-6. Design Characteristics for Example I

	VESSEL A		VESSEL B	
P_{meop}	100 psi	0.689 MN/m^2	1000 psi	6.895 MN/m^2
MATERIAL PROPERTIES[1]				
• F_{tu}	129 ksi	889 MN/m^2	129 ksi	889 MN/m^2
• F_{ty}	118 ksi	814 MN/m^2	118 ksi	814 MN/m^2
• K_{Ic}	92 ksi $\sqrt{in.}$	101 $\frac{MN}{m^2}\sqrt{m}$	92 ksi $\sqrt{in.}$	101 $\frac{MN}{m^2}\sqrt{m}$
DESIGN CHARACTERISTICS				
• Factors of Safety [2]				
- Safe-Life	1.84		(SATISFIED)	
- Fail-Safe	(SATISFIED)		3.81	
- Proof				
Initial	1.69		1.69	
Subsequent	1.24		1.24	
• Vessel Thickness [3]	0.013 in.	0.033 cm	0.266 in.	0.676 cm
• Surface Density	0.00368 psi	0.261 g/m^2	0.0753 psi	5.30 g/m^2
• Vessel Weight	14.5 lb.	6.54 kg	306 lb.	139 kg
• Critical Flaw Depth[4]				
- Proof Pressure				
Initial	0.063 in.	0.160 cm	0.266 in.	0.676 cm
Subsequent	0.114 in.	0.290 cm	0.493 in.	1.25 cm
- Operating Pressure	0.176 in.	0.447 cm	0.760 in.	1.93 cm
• Failure Mode	LEAK		LEAK	

NOTES: (1) From Table D-1.

(2) From Table D-4 or Figure D-13.

(3) Calculated from the equation $t = \frac{P_{meop} R \times F.S.}{2 F_{tu}}$

(4) Calculated from the equation $a_{cr} = \left(\frac{Q}{1.21\pi M_k^2}\right)\left(\frac{K_{Ic}}{\sigma}\right)^2$

with $Q = 1$ and $M_k = 1.6$.

- Uses: 15
- Operating Pressure, p_{meop} = 6.985 mn/m^2 (1000 psi)
- Configuration: Spherical pressure vessel volume requirement ~ 400,408 cm^3 (24,430 in.^{3})
- Material: T-1 Steel
- Environmental Effect: K_{TH}/K_{Ic} = 0.8

The Design Characteristics are summarized in Table D-7. It is seen that multiple-tank arrangement results in considerable weight savings when pressure vessels are designed to internal volume constraints. In this context, the total weight for the 10-tank configuration of 66.2 kg (146 lb) is substantially less than the single tank weight of 139 kg (306 lb). In all cases, the proof-test factors are identical to satisfy safe-life requirements, but the design factors of safety differ sharply. For this example problem, fail-safe requirements dictate the selection of the appropriate design factors of safety for all the multiple-tank arrangements except the 10-tank configuration. All the vessels satisfy the leak-before-burst criterion at the initial proof pressure levels and are fail safe against subsequent proof and operating pressure conditions. This example further illustrates that for weight-critical pressure vessels which must satisfy safe-life requirements and are required to ensure a fail-safe failure mode, the designer should resourt to multiple-tank arrangements whenever possible.

D. CONCLUSIONS

From the results of Tables D-2 to D-5, inclusively, and Figures D-11 to D-13, inclusively, the values of the proof factors and the design factors of safety are seen to be dependent upon the mechanical properties, fracture toughness, and flaw-growth rates of the materials, the mission requirements, the environmental effects, and the proof-test procedures. For expendabl pressure vessels, the design factors of safety for the six common alloys range from 1.20 to 1.33 when $K_{TH}/K_{Ic} \rightarrow 1.0$. These

Table D-7. Design Char

PHYSICAL DESCRIPTION	1		2
• Number of Tanks			
• Volume per Tank	24429 in^3	400,320 cm^3	12215 in^3
• Radius	18 in	45.7 cm	14.3 in
STRESS PARAMETER			
• $p_{meop}R$	18 ksi-in	3.15 MN/m	14.3 ksi-in
DESIGN CHARACTERISTICS			
• Factors of Safety *			
- Safe-Life	(SATISFIED)		(SATISF
- Fail-Safe	3.81		3.0
- Proof			
Initial	1.69		1.6
Subsequent	1.24		1.2
• Vessel Thickness *	0.266 in	0.676 cm	0.168 in
• Surface Density	0.0753 psi	5.30 g/cm^3	0.0474 psi
• Tank Weight			
- Each Tank	306 lb	139 kg	121 lb
- Total Weight	306 lb	139 kg	242 lb
• Critical Flaw Depth *			
- Proof Pressure			
Initial	0.266 in	0.676 cm	0.168 in
Subsequent	0.493 in	1.25 cm	0.311 in
- Operating Pressure	0.760 in	1.93 cm	0.478 in
• Failure Mode	LEAK		LE

* Obtained from same equations and source as Example 1, Table D-6.

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Table D-7. Design Characteristics for Example 2

	2		5		10	
320 cm^3	12215 in^3	200,160 cm^3	4886 in^3	80064 cm^3	2443 in^3	40032 cm^3
5.7 cm	14.3 in	36.3 cm	10.5 in	26.7 cm	8.35 in	21.2 cm
MN/m	14.3 ksi-in	2.5 MN/m	10.5 ksi-in	1.84 MN/m	8.35 ksi-in	1.46 MN/m
)	(SATISFIED)		(SATISFIED)		1.84	
	3.05		2.22		(SATISFIED)	
	1.69		1.69		1.69	
	1.24		1.24		1.24	
cm	0.168 in	0.427 cm	0.091 in	0.231 cm	0.060 in	0.152 cm
g/cm^3	0.0474 psi	3.34 g/cm^3	0.0257 psi	1.81 g/cm^3	0.0168 psi	1.18 g/cm^3
kg	121 lb	54.9 kg	35.8 lb	16.2 kg	14.6 lb	6.62 kg
kg	242 lb	109.8 kg	179 lb	81 kg	146 lb	66.2 kg
cm	0.168 in	0.424 cm	0.091 in	0.231 cm	0.060 in	0.152 cm
cm	0.311 in	0.790 cm	0.169 in	0.429 cm	0.113 in	0.287 cm
cm	0.478 in	1.21 cm	0.260 in	0.660 cm	0.175 in	0.445 cm
	LEAK		LEAK		LEAK	

mple 1, Table D-6.

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values are seen to agree with the conventional, although arbitrarily chosen, values of 1.25-1.40 for this type of pressure vessels. This agreement is particularly relevant due to the time-tested nature of the conventional values.

For the other types of Shuttle payloads, the values of design factors of safety differ considerably from the conventional values. The results of this investigation clearly demonstrate the inadequacy of the single factor of safety design philosophy practiced conventionally. In fact, no single fixed factor of safety is appropriate to ensure safe-life or fail-safe behavior for all pressure vessels, nor is the single factor of safety design philosophy adequate to address the diversified design possibilities offered by the increased capability and versatility of the Space Shuttle program. By suitable application of the principles of fracture mechanics, candidate materials for specified mission objectives can be ranked on the basis of their structural performance, resulting in improved harmony among materials, design requirements, and safety of airborne pressure vessels. In addition, meaningful design tradeoffs are possible when factors of safety are established in a rational manner.

E. RECOMMENDATIONS

It is recommended that this guideline be enforced on all pressure vessel design for the four types of Shuttle payloads under consideration. To further quantify the values for the proof and design factors of safety, it is also recommended that a detailed material characterization program be implemented so that specific flaw-growth rates, strengths and fracture toughness date, etc. in the expected operating environments can be ascertained for candidate pressure vessel materials in the appropriate fluid environment.

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APPENDIX E. THERMAL CONTROL

A. INTRODUCTION

The basic function of the thermal control subsystem is to provide thermal control to the other subsystems within the payload. With the potentially greater payload weight and volume capability available in the Shuttle era, the thermal control ubsystem (TCS) should be designed so that there are no payload failures caused by loss of thermal control.

In order to effect this, tradeoff studies should be performed to identify the penalties associated with various degrees of cold side bias. Cold side bias requires designing the TCS to operate at or near the cold end of a subsystem/component's acceptable temperature range. With this type of design approach, the result will be a subsystem/component with better reliability characteristics because of its lower operating temperature and greater temperature margin on the hot side. The cold side temperature margin which is lost by this design approach will be recouped by adding heaters. The primary tradeoff involves the additional weight and volume associated with the heater power versus the increase in mean mission duration (MMD) and increased confidence in thermal control performance.

In the modular spacecraft concept, each module should have its own autonomous TCS. This will alleviate interface problems such as fluid connects and mating of critical heat paths.

Generally, subsystems/components have different temperature limits for operating and non-operating conditions. The use of a cold side bias design is directed toward the operating temperature limits. Consideration will need to be given to designing the TCS so that subsystem/component temperature can be passively controlled when the normal TCS system is inoperable. This will provide capability to maintain acceptable temperatures during completely quiescent modes.

B. ANALYSIS

1. COST CONSIDERATIONS

The objective of this appendix is to provide guidelines for the production of low cost, reliable payloads launched with the Shuttle. Cost is distributed over a large fleet of spacecraft and not charged to just one specific vehicle. In order to realistically assess the cost savings potential, a cursory review of subsystem relative costs was made. Cost information was taken from Reference E-1 and is shown in Table E-1 to provide an appreciation of the relative costs of spacecraft subsystems. The data are for four different representative NASA spacecraft, and both RDT&E and recurring costs are shown.

Experiment costs are not included in this table. Although the percentages for the various subsystems differ among the four missions because of differences in orbital and mission objectives, the fact that thermal control costs are relatively small is irrefutable. Therefore, reducing costs directly related to thermal requirements will not significantly reduce the cost of the spacecraft. Instead, additional uncertainties which may be introduced into the thermal analysis by reducing the total thermal effort may result in increased costs due to failures of the vehicle.

The basic purpose of the thermal control subsystem is to provide a support function. It ensures a benign environment in which the other subsystems may function properly. Failure of the thermal control subsystem may result not only in loss of the thermal control subsystem proper but also in the failure of one or more of the other subsystems.

2. REVIEW OF SPACECRAFT ANOMALIES

The extent to which thermal problems currently result in spacecraft anomalous behavior will have some bearing on the approaches taken for a Shuttle payload. If thermal anomalies are shown to be minimal and result in minimal impact on spacecraft performance, the thermal design approach probably will not differ in the Shuttle era from what is done today.

Table E-1. Percentage Costs of Payload Subsystems

SUBSYSTEM	ORBITING ASTRONOMY OBSERVATORY		MARS ORBITER		SMALL RESEARCH SATELLITE		SYNCHRONOUS EQUATORIAL ORBITER	
	RDT&E	RECURRING	RDT&E	RECURRING	RDT&E	RECURRING	RDT&E	RECURRING
STRUCTURES	5.90	18.38	14.77	13.54	9.93	4.09	14.39	19.08
ELECTRICAL & PYRO	10.35	12.55	17.81	19.34	40.53	42.59	18.67	24.91
STABILIZATION & CONTROL	50.04	45.72	16.29	25.14	1.16	2.73	16.29	16.49
PROPULSION			3.26	3.22				
ATTITUDE CONTROL	2.17	1.03	3.94	2.15	4.55	8.52	4.21	3.00
CDPI	27.40	18.96	43.45	36.10	41.36	38.33	46.19	35.92
ENVIRONMENTAL CONTROL	4.14	3.36	0.47	0.52	2.48	3.75	0.24	0.61
TOTAL	100	100	100	100	100	100	100	100

Failures and anomalies which occurred on selected Air Force Programs were reviewed to determine any common underlying causes and to determine corrective actions (Ref. E-2). Failures/anomalies were classified by discipline, and the results show that of the failure/anomalies reviewed, 11.6 percent were attributable to thermal design. Failures/anomalies included overheating caused by surface degradation or contamination, thermal runaway of batteries, and inaccuracies in temperature predictions due to analysis and simulation. Although the 11.6 percent failures/anomalies did not result in total loss of mission, thermal subsystem failures/anomalies did cause a disportionately higher percentage of anomalies than its cost would indicate.

The basic finding of the failure/anomaly review was that a significant number of failure/anomalies were due to inadequate thermal design resulting from insufficient temperature margins in the TCS design. It must be concluded that where system tradeoffs dictate, greater margins should be included in the TCS design. Cold side bias with active heaters is a means of providing greater temperature margins.

3. EFFECT OF TEMPERATURE ON FAILURE RATE

It has been shown that for semi-conductors and microelectronic integrated circuits, the Arrhenius model gives a fairly good representation of the failure rate (Ref. E-3). The Arrhenius model states that the failure rate is exponentially related to the negative of the reciprocal of the absolute temperature. Figure E-1 illustrates this relationship.

Mechanical failures, such as breaking of solder joints and seals, are primarily caused by thermal stresses, and occur more readily at lower temperatures as shown in Figure E-1. The resultant total failure rate also is shown as a dashed line.

- EFFECT OF TEMPERATURE ON FAILURE RATE

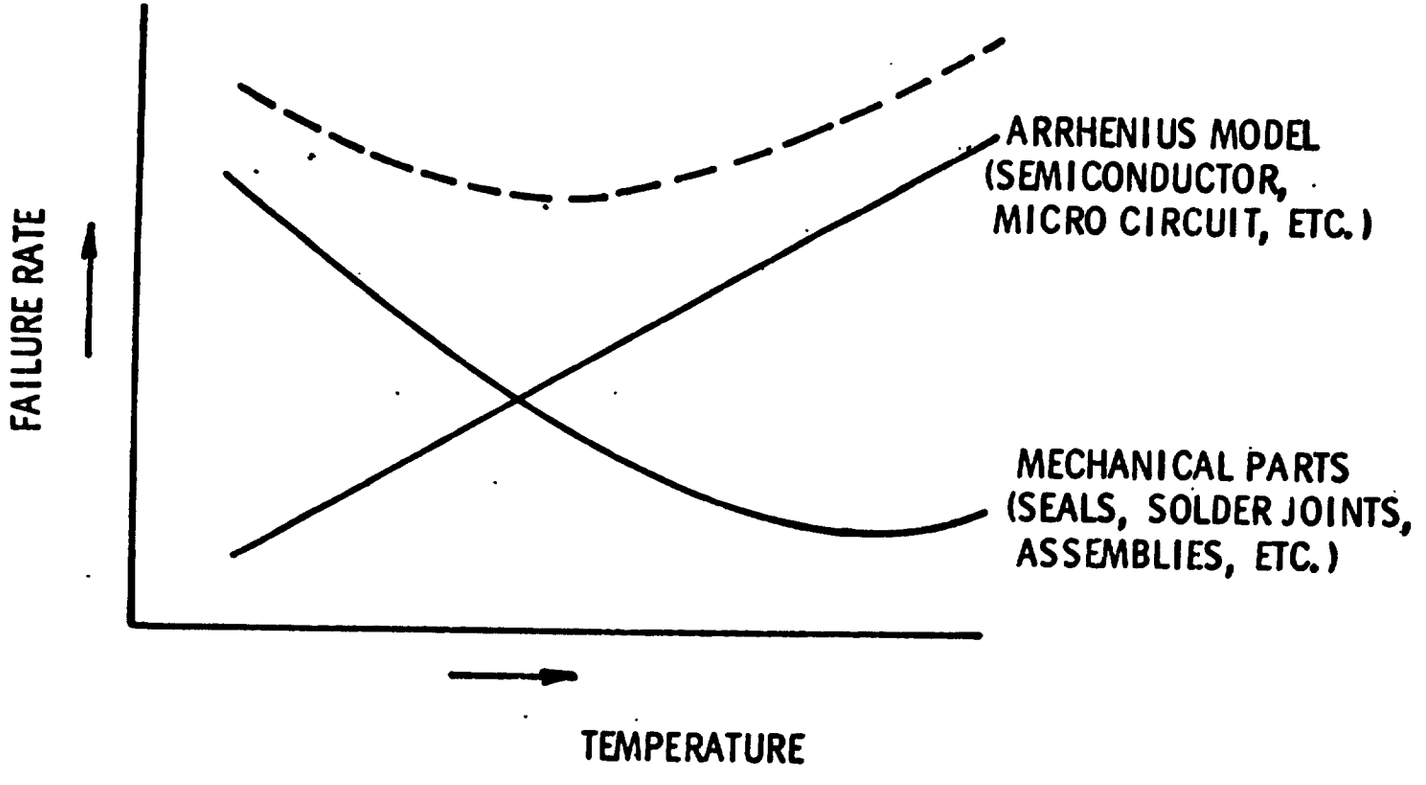


Figure E-1. Thermal Design

Another consideration is the effect of temperature variation or temperature cycling on failure rate. The effect of temperature cycling is not well known in terms of quantitative failure rates, but qualitatively, the failure rate increases with increasing temperature cycling.

Although the quantitative effect of temperature on failure rate is not well defined because of the individual characteristics of each component and despite the dearth of available data of this type, several general conclusions can be drawn regarding the effect of temperature on failure rate:

a. Failure rate will increase with both too low and with too high an operating temperature.

b. There is generally a minimum in the temperature-failure rate curve.

c. Temperature cycling tends to increase failure rate.

Thermally, the message is that in order to reduce temperature-induced failure rates, components should be run at a relatively constant temperature. The temperature level at which a specific component operates optimally (from failure rate considerations) is characteristic of each specific component. Typically, however, optimal temperatures are near the lower end of the currently utilized temperature ranges. Although documented data is meager supporting the fact that the lower end of the temperature range is near optimal, data on NiCd batteries was found which supports this contention. (Ref. E-4).

4. COLD SIDE BIAS

Common practice is to design the thermal control subsystem so that there are equal margins on both the hot and cold sides. The rationale is that it is equally difficult to predict hot or cold temperatures, and a component is therefore equally likely to fail as a result of either a too cold or a too hot environment.

Cold side bias is a concept wherein a component is designed thermally to operate relatively near its cold temperature limit. The result is that additional temperature margin is gained on the hot side with loss of temperature margins on the cold side. Heaters are then used to provide the necessary cold side margin.

The cold side bias concept is illustrated in Figure E-2 which shows internal heat flux as a function of temperature for both the maximum and minimum environmental conditions for a typical geosynchronous orbit. A typical electronic equipment operating range is also shown in Figure E-2.

When operating in the cold bias mode, the nominal temperature of the equipment is reduced, and the temperature margin on the hot side is increased. These advantages are not free; the penalty to provide these characteristics is additional heater power. The amount of additional power required is dependent on the orbital conditions and the amount of heater margin one wants to incorporate into the design. The additional power requirement will probably be acceptable because of the large weight and volume capability of the Space Shuttle.

Figures E-3 through E-5 show radiator heat rejection capability as a function of temperature for various orbits. The minimum and maximum environments refer to the minimum and maximum orbital heating conditions. For example, for a radiator with a constant internal load, its temperature will cycle (horizontally) between the minimum and maximum environments. Thermal extremes are greatest in the synchronous orbit and smallest for the low earth orbit radiator facing the earth.

5. EFFECT OF COLD SIDE BIAS DESIGN ON SURFACE PROPERTY CONSIDERATIONS

Past experience has shown that thermal surface property degradation results in substandard thermal performance of certain spacecraft. The primary causes for performance degradation have been UV radiation damage and surface contamination. The result of these types of

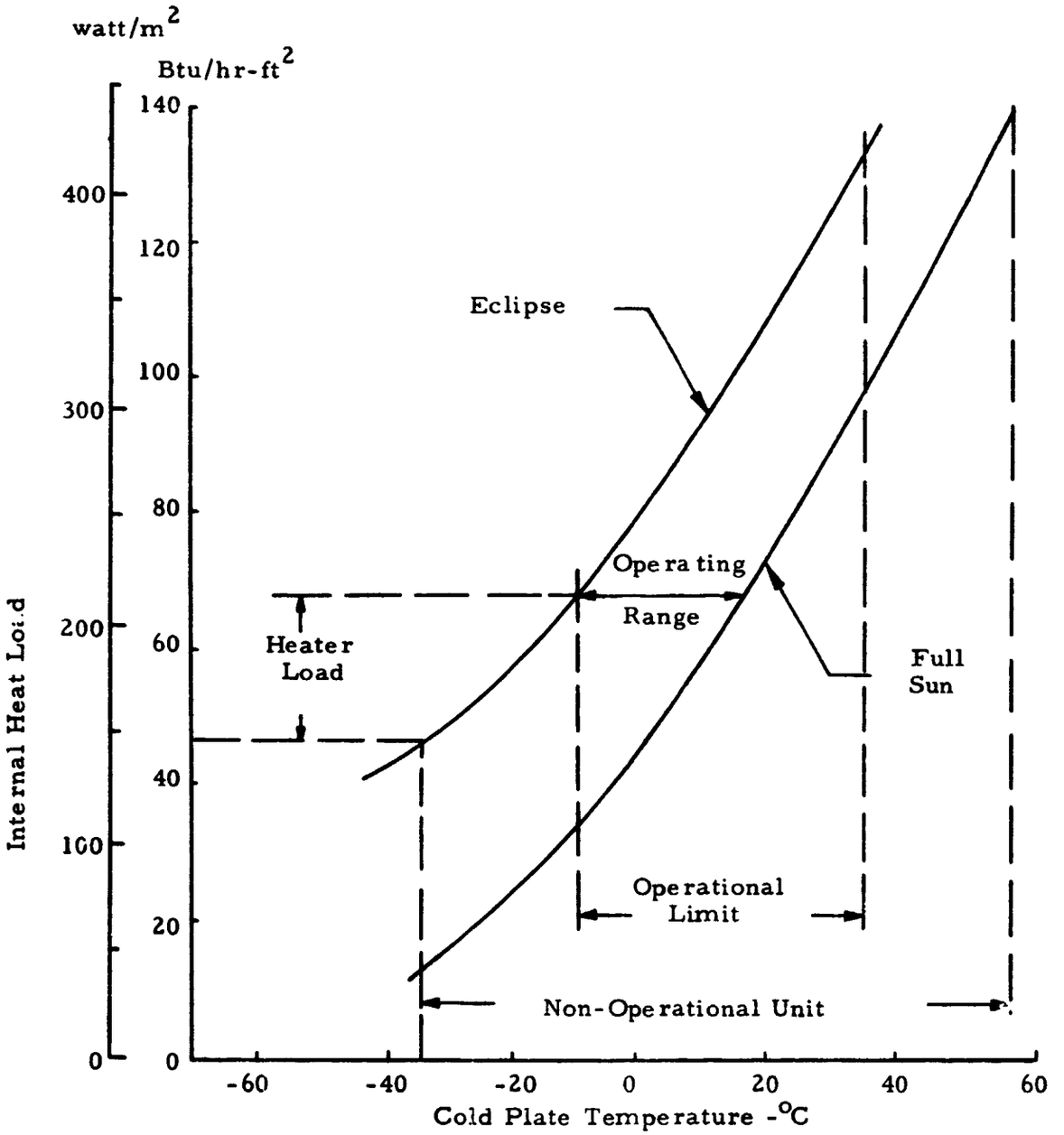


Figure E-2. Cold Side Design Concept

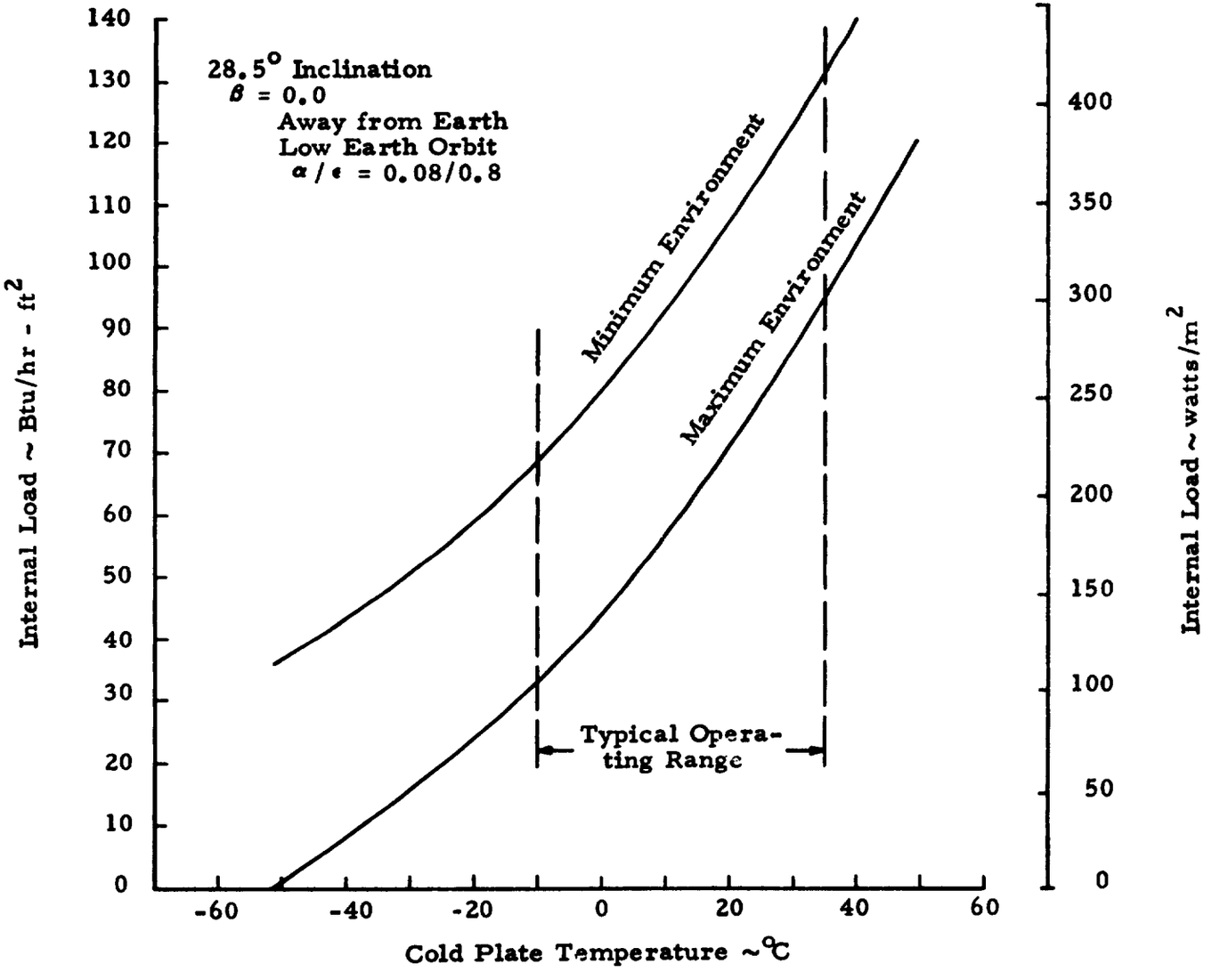


Figure E-3. Cold Plate Heat Rejection Versus Temperature

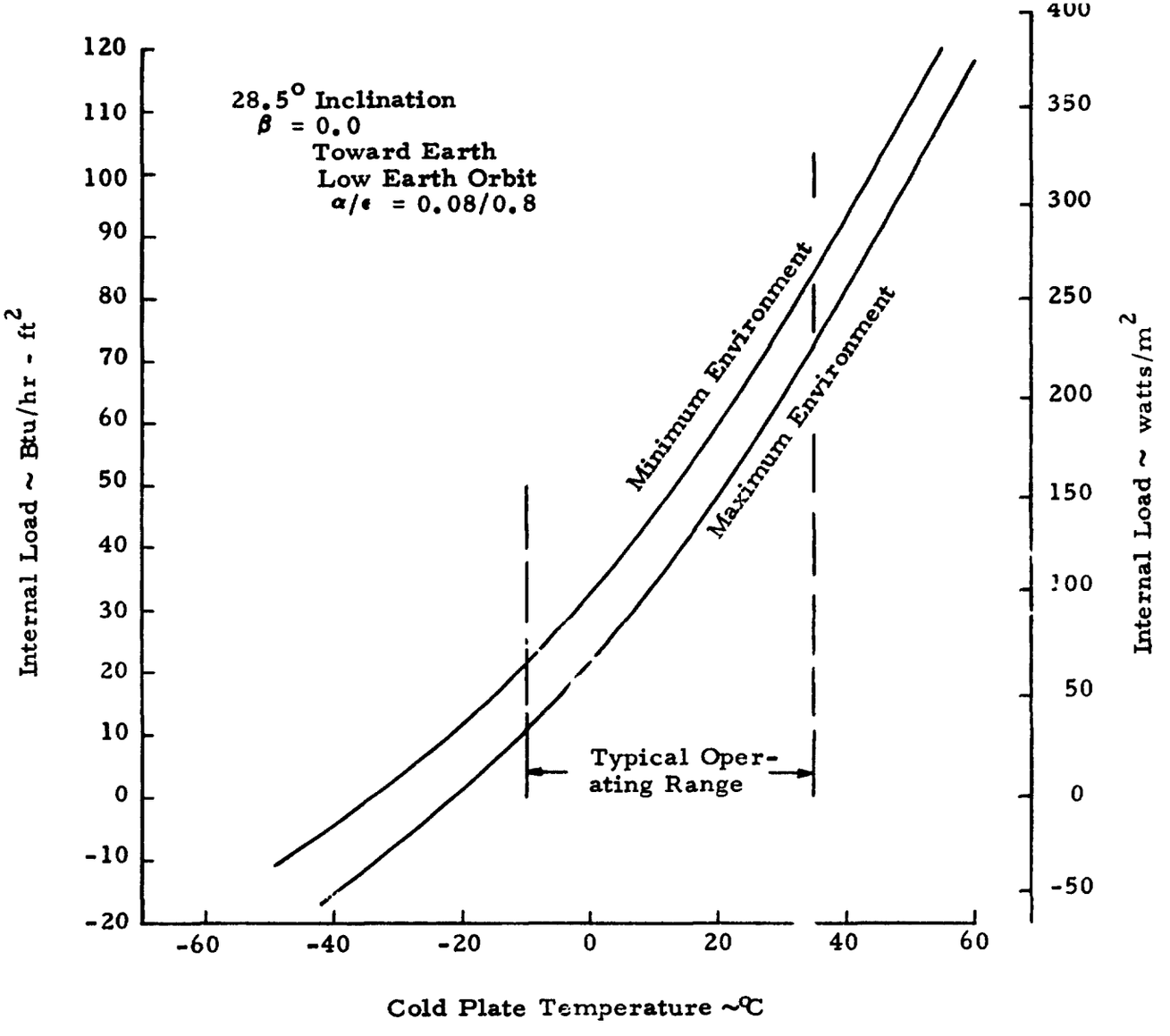


Figure E-4. Cold Plate Heat Rejection Versus Temperature

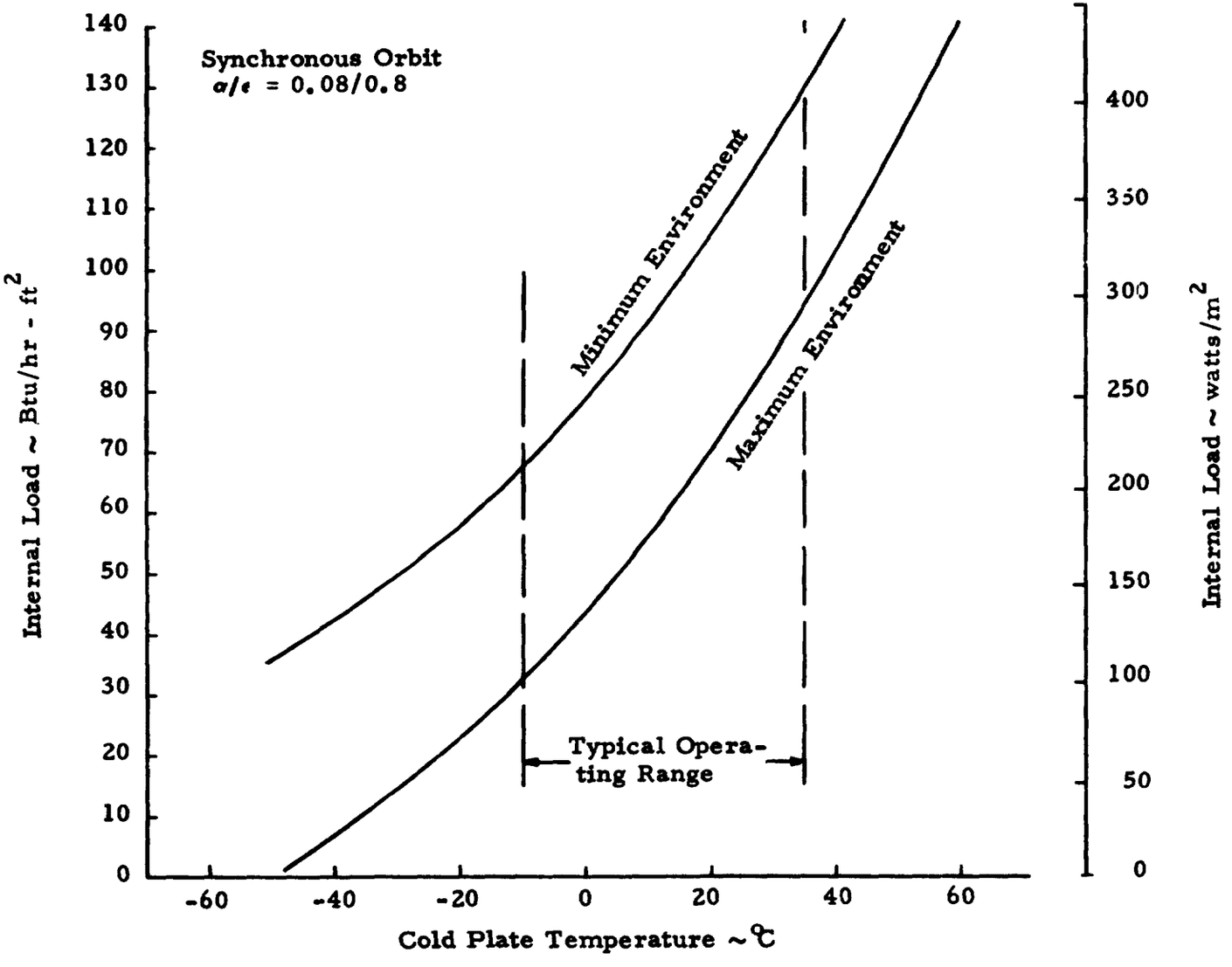


Figure E-5. Cold Plate Heat Rejection Versus Temperature

damages is primarily an increase in surface absorptivity. One spacecraft using second surface mirrors is showing an α degradation of approximately two percent per month, and similar degradation data has been shown for white paints (Ref. E-5). Consequently temperatures increase.

The use of the cold side bias concept for common spacecraft equipment results in increased temperature margins on the hot side. If the causes of solar absorptivity increase cannot be alleviated by design or material control consideration, nominal temperatures can be expected to increase. However, the cold side bias concept increases the available hot side temperature margin so that the time to reach the maximum permissible temperature, and therefore vehicle life, is increased.

6. PASSIVE THERMAL CONTROL IN THE NON-OPERATIONAL MODE

The use of a cold side bias design will result in a relatively constant temperature component design at a relatively low temperature. The use of a cold side bias design concept, however, will have limitations due to several considerations. First, the orbital considerations and the tradeoff between the benefits of cold side bias and heater power must be considered. Second, the problem of a module or subsystem getting too cold in a failed mode must be considered.

In present expendable satellite design, no consideration is given to the design requirements for a salvageable satellite in the event of a failure. When one speaks of refurbishing a satellite, one must consider maintaining the failed satellite in a refurbishable condition. Additionally, in the event of a failure, the design must effectively isolate the failure so that secondary failures will not result. In the area of thermal control, the primary cause of secondary failures is components becoming excessively cold after a power failure. A power failure may result in both the loss of internally generated heat and heater power capability. Subsequently, the component may be permanently damaged by the excessive cold. In order to avoid this problem, the use of the cold side bias design must be tempered

with the constraint that the TCS should be required to prevent component temperature in a zero-powered condition from falling lower than the component specified minimum non-operating temperature.

7. APPLICABILITY TO PAYLOADS

The philosophy of improving TCS performance to provide better payload temperature control is applicable to expendable, on-orbit maintainable, and ground refurbishable payloads. The extent to which cold side bias can be applied is dependent on the weight and power penalties involved. The acceptability of these penalties is rather a function of the orbit than the type of mission. Orbital parameters are important, because they define the space environmental condition and also determine the payload allowable total weight.

C. CONCLUSIONS

In determining thermal guidelines for spacecraft design in the Shuttle era, several initial considerations were investigated. First, since the ultimate objective is to provide guidelines for a low cost payload, spacecraft subsystem relative costs (excluding experiment costs) were reviewed. Second, spacecraft experienced failures and anomalies were reviewed to determine the extent to which thermal problems have contributed to failures/anomalies in the past. Third, the effect of temperature on failure rate was evaluated. Finally, a thermal guideline was developed which is believed will result in a more reliable and cheaper vehicle.

A review of spacecraft system cost data revealed that thermal subsystems were one of the least expensive subsystems. Thermal control amounted to less than five percent in each of four typical NASA spacecraft missions reviewed. Consequently, a significant cost savings will not be effected by reducing TCS related costs. Even if TCS related costs can be totally eliminated, total cost will not be reduced appreciably. Consideration of spacecraft experienced failures and anomalies seems to indicate that in

the Shuttle era, more effort (both analytical and experimental) should be expended in the area of thermal control, because the results of failures/ anomalies attributable to thermal design may result in losses far greater in cost than the actual cost of the TCS.

It would appear that the thermal design approach to use in the Shuttle era would be to incorporate an improved TCS to increase performance and reliability. The primary obstacles to improved thermal design traditionally have been constraints on vehicle weight and volume. Weight limitations are reflected primarily in constraints on available power for thermal control. With the Shuttle, it is anticipated that weight and volume constraints will be relaxed to result in a better spacecraft design.

Reliability and failure rate data indicate that temperature has an impact on MMD considerations. First order considerations indicate that electronics failure rates can be approximated by Arrhenius' Law, which states that failure rate is exponentially related to the reciprocal of the absolute temperature. Consequently, it appears that from a component/ subsystem reliability standpoint, it will be advantageous to operate the component at a reasonably low temperature. However, the lower temperature limit will be limited by mechanical-failure-rate considerations. Additionally, the implication is that it is advantageous to operate components at a relatively constant temperature.

A basic conclusion which was deduced from the above review was that trying to reduce thermal cost will not significantly reduce program cost. Additionally, since thermal control is a relatively low cost item, the additional weight and volume available for Shuttle payloads should be used to provide a better performing TCS which should result in a better performing spacecraft.

These conclusions and reliability data resulted in the proposed application of the cold side bias concept in which components are operated near the cold end of their thermal operating range. In order to do this, heater power is required.

D. RECOMMENDATIONS

It is recommended that in the design of Shuttle payloads, the additional weight and volume capability of the Shuttle be utilized in part to provide more stringent temperature control in order to increase subsystem/component performance characteristics. Although this may result in a heavier and perhaps more expensive thermal control subsystem, it is believed that on an overall systems basis, it will result in a cheaper vehicle because of improved performance, life, and reliability.

The degree to which better temperature control can be effected will depend on the penalties involved (weight, power, volume). Consequently, it is also recommended that improved methods of temperature control be evaluated on an individual payload basis because of the differences in the importance of tradeoff parameters for each payload.

It is further recommended that cold side bias be considered as a means of providing improved thermal control.

E. REFERENCES

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APPENDIX F. REACTION CONTROL

A. INTRODUCTION

The development and fabrication of high-efficiency, reliable reaction control subsystems for conventional satellites has been (historically) very costly. This situation is the result of the limited launch weight capacity of expendable boosters which imposes stringent weight limits on the satellite, high reliability demands, and highly precise impulse bits required by the satellite control subsystem. It is apparent that the unique capabilities of the Space Shuttle (excess launch-weight capacity, orbit rendezvous, and retrieval) may be utilized to substantially reduce these costs while maintaining the required satellite operational reliability.

B. ASSUMPTIONS

The reaction control subsystem provides thrust for such functions as attitude control, stationkeeping, spin-up, despin, on-orbit maneuvering, spin-axis precession, and apsis corrections. This analysis specifically excludes the orbit injection maneuver which requires its own unique tradeoff study comparing the use of the Shuttle/Tug, an expendable kick stage, or an additional propulsion subsystem within the payload.

Table F-1 lists some of the types of propulsion subsystems available for consideration in payload design. For this analysis, it is assumed sufficient to analyze only the three most common types, which are inert gas, hydrazine monopropellant, and the N_2O_4/MMH bipropellant.

Table F-1. Types of Propulsion Systems Considered

Inert Gas

- Freon
- *Nitrogen
- Helium
- Argon
- Ammonia

Mixed Gases

- Tridyne ($0_2 + H_2 + N_2$)
- Freon - Nitrogen
- Helium - Nitrogen

Liquid Monopropellant

- *Hydrazine
- Peroxide

Subliming Solid

Solid Propellant

Liquid Bipropellant

- *Storables ($N_2 0_4$/MMH, IRFNA/UDMH, etc.)
- Cryogenics (F_2/H_2, $0_2/H_2$, etc.)

Electrolysis

- Hydrazine
- Water
- Peroxide

Heated Gas

- Electrical (N_2, NH_3, etc.)
- Radioisotope (Hg, C_5, NH_3)
- Nuclear Reactor (H_2)

Ionic

- Cesium

Pulsed Plasma

- Teflon
- Mercury

Colloid

- Glycerol

* Most commonly used for satellites

In order to reduce study examples to manageable numbers of variables, it is assumed that the payload requires 12 thrusters in the 89-N (20-lb) or less thrust range. Assuming specific impulses of 65 sec, 200 sec, and 300 sec for nitrogen, hydrazine monopropellant, and bipropellant, respectively, the propulsion subsystem weight, cost and volume were determined and are presented graphically in Figures F-1 through F-3. These results are useful for comparative purposes during preliminary design and as an example of the approach to an appropriate system tradeoff analysis.

C. ANALYSIS

The approach to selection of the most cost-effective propulsion subsystem is keyed to the payload type. An expendable payload with a long life may require a much more sophisticated system than an on-orbit maintainable or ground-refurbishable payload in order to meet the weight and reliability requirements. However, the first consideration should be a pressure-regulated nitrogen gas subsystem. It is the simplest and least costly of any of the subsystems for total impulse requirements less than approximately 6.7×10^4 newton sec (15,000 lb sec). As requirements exceed this value, the inert gas subsystem becomes too costly, primarily due to the large volume, high pressure tankage required. Comparative weights of the three systems versus total impulse requirements are given in Figure F-1, and the rapid increase in weight with total impulse requirements of the nitrogen subsystem is apparent here. For a total impulse delivered of 6.7×10^4 newton sec (15,000 lb sec), the inert gas subsystem is four times as heavy as the other two subsystems, and the tank volume is an order of magnitude greater than that for the liquid subsystems (Figure F-3). An assumption of "no redundancy" was made in the data provided in Figures F-1 through F-3. The data do not include cost or weight of any positive expulsion device. Because cost

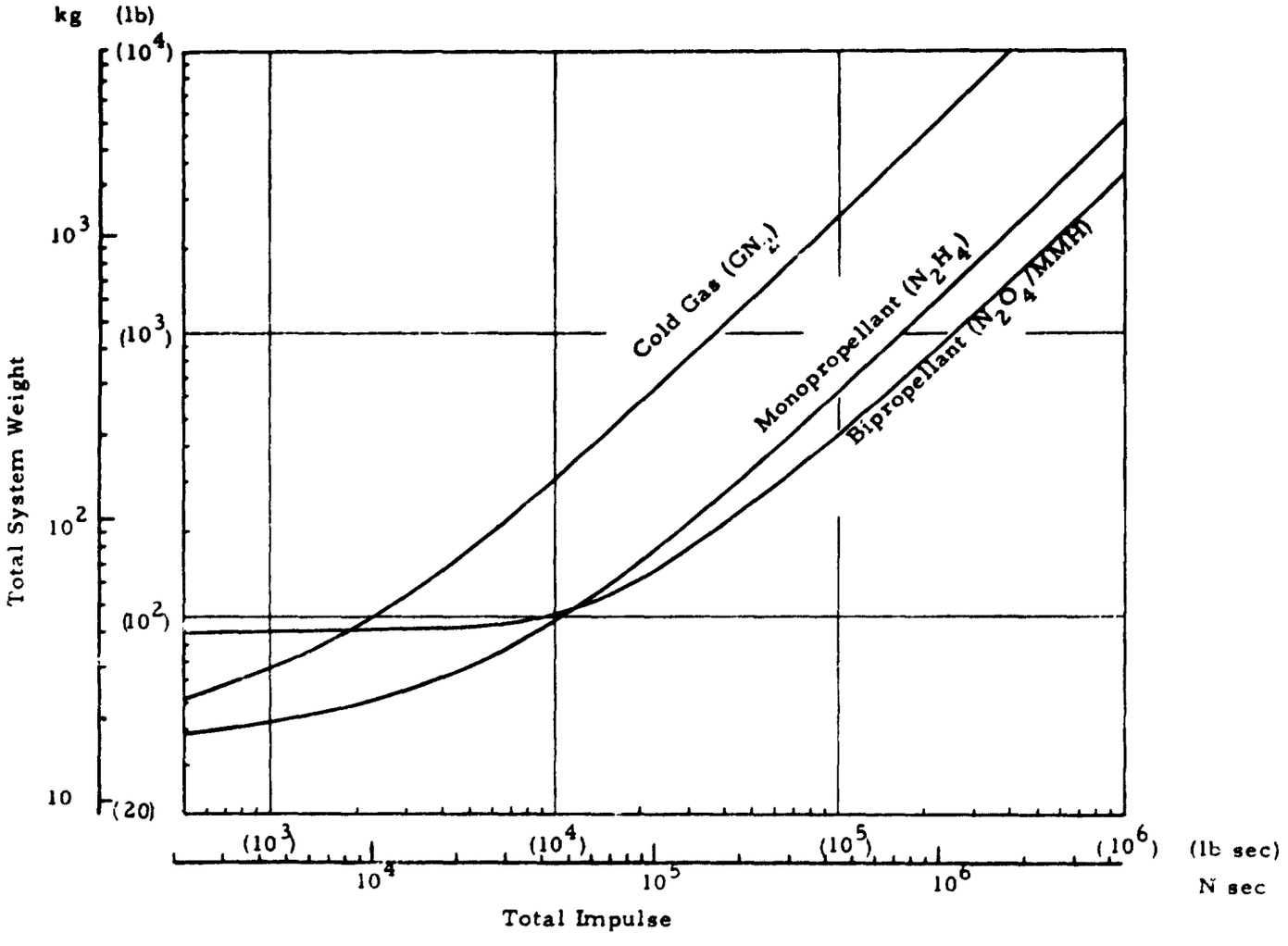


Figure F-1. System Loaded Weight as a Function of Total Impulse and Propellant Type

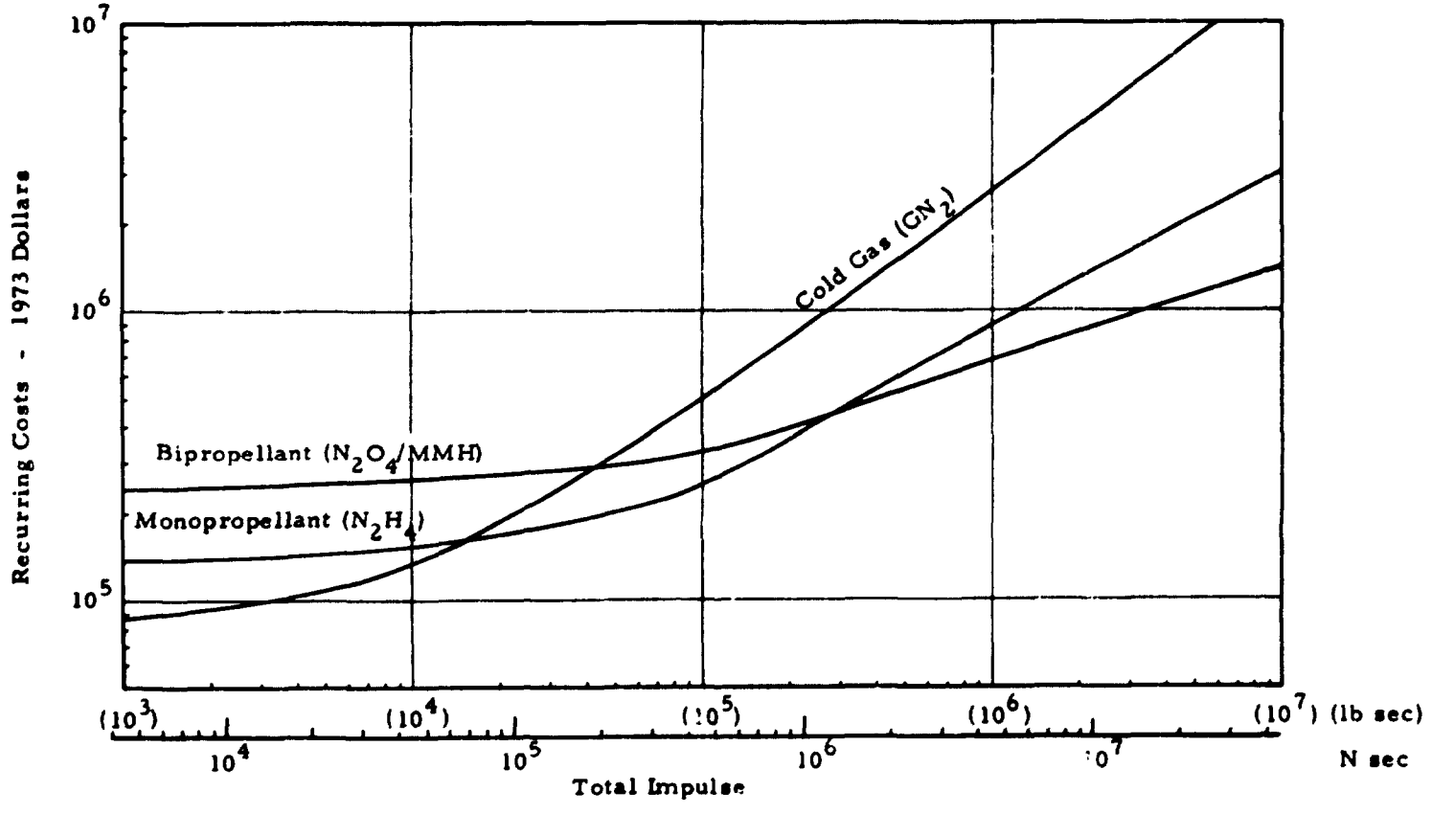


Figure F-2. Recurring Procurement Costs as a Function of Total Impulse and Propellant Type

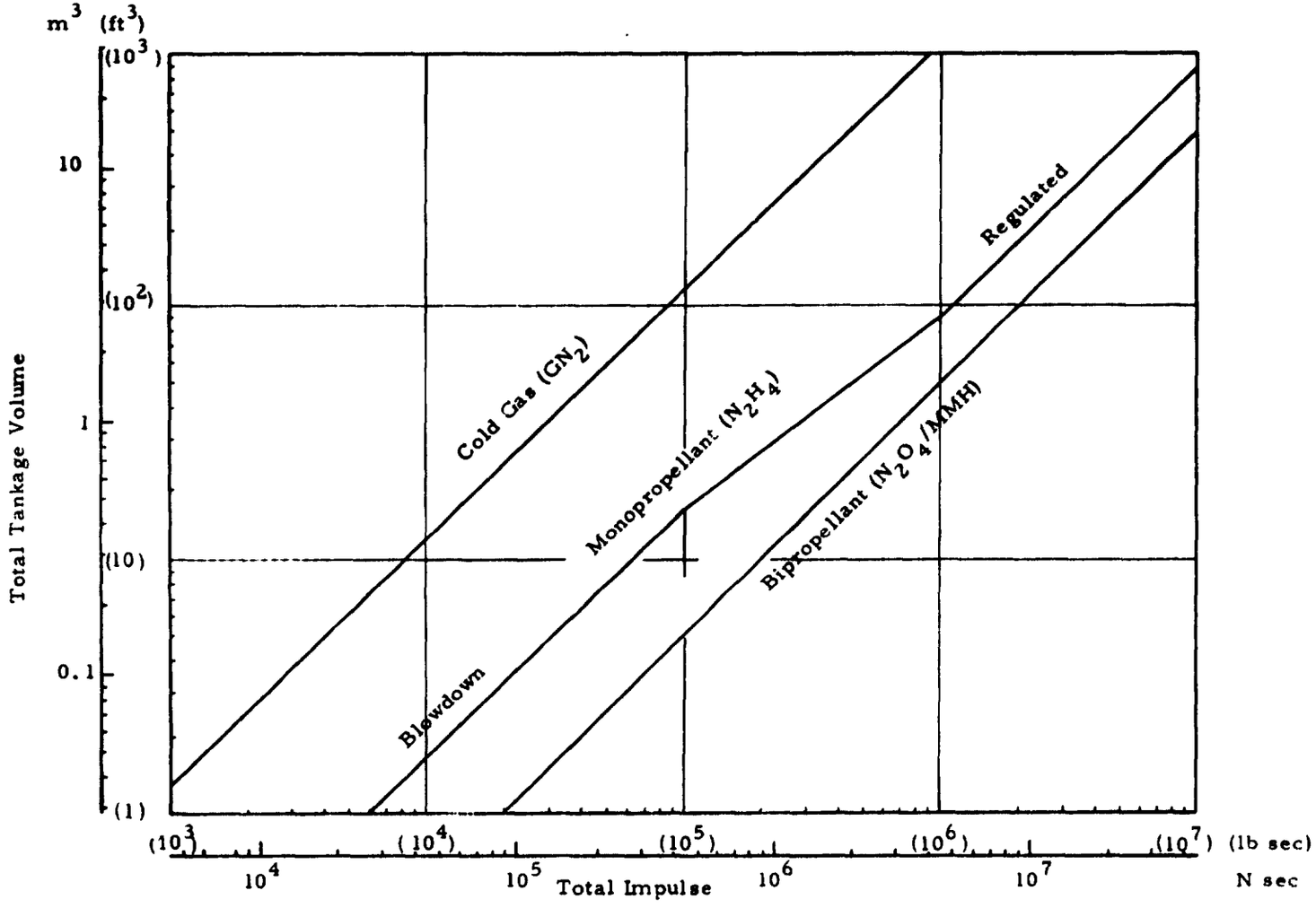


Figure F-3. Volume (Propellant Plus Pressurant Tankage) as a Function of Total Impulse and Propellant Type

is the prime consideration, the nitrogen gas subsystem should be given first choice if the weight and volume are not incompatible with system requirements.

If the nitrogen subsystem is not feasible (because of high total impulse requirements or excessive weight), the next consideration should be the hydrazine monopropellant subsystem. A typical N_2H_4 subsystem would operate in the blowdown mode (no regulator) with the propellant tank filled with hydrazine and pressurant in approximately equal volumes. Such a subsystem actually has fewer components than an inert gas system and may have equivalent or higher reliability is will depend on the mission duration/refurbishment cycle in which the expected failure rate of mono-propellant thrusters is traded against the failure rate of a regulator.

The hydrazine subsystem shows a cost advantage over a bipropellant subsystem for total impulse requirements less than 13.3×10^5 newton sec (300,000 lb sec). At this point, the greater volume and weight of the hydrazine propellant required demand large costly tank volumes. Also, in this region it becomes necessary to consider a pressure-regulated subsystem to reduce system volume by storing the pressurant in smaller high pressure tanks, thereby adding complexity and grading reliability.

If neither of the discussed subsystems can meet the total impulse requirements within the weight and volume constraints, the liquid bipropellant subsystem remains as the only choice. Here the tradeoff is between the reduced weight and volume of a bipropellant subsystem against its increased complexity and hazard and reduced reliability.

A mention should be made of a combined mono/bipropellant subsystem using N_2O_4/N_2H_4 propellants. In this concept, hydrazine

can be used to power small attitude control thrusters which operate mostly in a pulsing mode, while the bipropellant with high specific impulse can be used for any larger engines expending a greater amount of total impulse.

For the liquid propellant subsystems, operation under zero g environment will be a requirement. To meet this requirement, bladders or surface-tension devices will be employed. Bladders may be either organic materials or metals. The selection will be made on the basis of material compatibility with the fluids used and the number of expulsion cycles and temperatures to which they are subjected. In general, nonmetallic bladders are suitable for multi-expulsion cycle application while metallic ones are limited usually to one or two expulsions (exceptions to these are the metal bellows). Surface-tension devices have been successfully employed with large tanks on the tank's outlet to trap enough liquid to allow the start of a rocket engine (as on the Agena). They have also been experimentally demonstrated in smaller tanks but have not been used in the numbers that bladders have. Care must be exercised in the design of surface-tension devices to ensure that heat soakback does not vaporize the propellant downstream of the surface-tension screen creating a ΔP that forces the trapped liquid back through the screen during a coast period, leaving only gas between the screen and the rocket engine inlet.

For some specialized applications, the water electrolysis propulsion subsystem could be considered. This subsystem is applicable to extended-duration missions that do not require large intermittent expenditures of propellant which would require prohibitively large electrical power and/or gaseous (hydrogen and oxygen) storage. This type of propellant generation (the evolution of hydrogen and oxygen from water by electrolysis) has been demonstrated but has not been developed to the point of flight-qualified hardware. The inherent advantages of the system are high I_{sp} (combustion of H_2 and O_2), storage of the propellant as water (compact), very low hazard, no toxicity, etc.

The propulsion subsystem components can be divided into three categories with respect to their expected reliability. These components are grouped in Table F-2. The Category I components are low reliability and have a useful life limit expressed in number of cycles. The low reliability of propellant tanks is due to the corrosion hazard of liquid propellants. Increased safety margins for such tanks will be an important factor in improving their reliability. The components in Category II are of medium reliability, are, in general, not subject to mission time limitation, and could be reused in a refurbishment cycle after a suitable checkout test. The components in Category III have essentially unlimited life. It is possible to bring the high-pressure vessels into that category if the safety factors are suitably increased. The costs shown are typical for small number production components and do not reflect the cost reduction which could be realized with larger production batches and modified specifications.

A detailed discussion of the properties of the three types of interest is presented in the following paragraphs along with the rationale for the RCS design approach for each type.

1. EXPENDABLE

The selection of a propulsion subsystem for an expendable payload is mainly mission dependent. The impulse requirements, the life on orbit, reaction force levels, etc. are determined by the mission, satellite mass properties, and attitude control method employed. The propulsion selection then is made to provide the reaction control subsystem that will meet the required mission reliability with minimum cost.

The tradeoff of weight, cost, and volume for the reaction control subsystems that will meet the technical requirements of the mission must be made (as illustrated in Figures F-1 through F-3). The selection will then be self evident. In determining projected reliability levels, the factors listed in Table F-2 may be used as a guide.

Table F-2. Typical RCS Component Characteristics

Component	Typical Cycle Life	Typical Production Cost ($)	Typical Reliability Factor
Category I			10^{-9}/hr
Propellant Tanks	50	8,000 - 16,000/ft^3	330
Thrusters, N_2H_4	100,000	4,000 each	166
Thrusters, N_2O_4/MMH	50,000	8,000 each	400
Thruster Valves GN_2	200,000	4,000 each	347
Thruster Valves, N_2H_4	100,000	4,000 each	347
Thruster Valves, N_2H_4/MMH (2 req'd)	50,000	4,000 each	347
Pressure Transducers	100,000	500 each	340
Pneumatic Regulators	100,000	5,000 each	245
Check Valves	100,000	500 each	250
Positive Expulsion Bladders			300
Category II			
Nitrogen Tank	50	4,000 - 5,000/ft^3	14
Thrusters, GN_2	200,000	500 each	17
Fill and Drain Valves	50	500 each	70
Thermisters		500 each	25
Category III			
Pyro Valves	1	500 each	---
Filters	1	500 each	1-10
Piping & Brazed Joints	---	---	1-10
Brackets, Structural Components	---	---	1

In principle, the expendable payloads do not require propulsion modularization since no refurbishment is possible, although in pre-orbital checkout in the Shuttle, a replacement of a module, if found defective, could enhance the overall mission success.

However, if the principle of standardization of certain subsystems, including propulsion, is developed because of the on-orbit or on-ground refurbishable payloads, it may be economically attractive to use similar standard housekeeping modules in expendable payloads.

The question of redundancy of propulsion components for expendable payloads should be resolved by consideration of component reliability, such as shown in Table F-2, against the desired mission time and reliability goals.

A number of options are open to the designer to achieve minimum cost at the required system reliability level. If the total impulse requirements are low, a totally redundant nitrogen gas subsystem might be less expensive and more reliable than a single hydrazine subsystem. Lower reliability components within a single hydrazine subsystem may be made redundant to improve its reliability. There may be, indeed, a great need for redundancy in this type of payload as opposed to the refurbishable type, particularly for long duration missions.

Consideration must be given to the Shuttle interface requirements for crew safety, propellant dump requirements (in case of an abort), environment, etc., and this is equally applicable to all of the other payloads.

2. ON-ORBIT MAINTAINABLE

The concept of maintaining a satellite on orbit as compared to the expendable concept may profoundly influence the design of the propulsion subsystem and the choice of propellants, since the choice of a subsystem based solely on the total impulse requirements may no

longer be valid. As an example, consider the long-life expendable satellite with large total impulse requirements that normally would force the choice of a bipropellant liquid subsystem to meet the weight and impulse limitations. If the same satellite may be visited some number of times during its life on orbit, a simple nitrogen gas subsystem that is resupplied at intervals may meet the system requirements at significantly lower cost and with higher reliability. Obviously, if the resupply missions should be flown exclusively to resupply the propellant, the cost of the resupply mission must be included in the tradeoff study and might negate the lower cost of the inert gas subsystem. It is assumed, however, that programmed on-orbit maintenance will be for the benefit of many systems and that the propulsion subsystem will be planned to use this available maintenance to best advantage.

The planning of the quantity of propellant and type, then, is to meet the impulse requirements between resupply visits. Another fundamental decision faces the designer at this point. Having selected the propellant, shall the resupply vehicle refill the original tanks on the satellite, replace tank modules with fully charged tank modules, or perhaps replace the complete propulsion subsystem modules which will contain a block of thrusters, tanks, and appropriate valving? The primary advantage for the last approach is that it eliminates propellant and pressurant interfaces (and potential leaks) and reduces interfaces to a single electrical connection for propulsion control. The disadvantage of this approach would be in the duplication of tanks, regulators, valves, etc., since two or more modules will be contained in a payload which increases the weight and cost. Also, such modules must be designed to carry the maximum propellant that is required for any situation (no propellant crossfeed capability), thus adding weight. Each module would be identical in order to minimize cost.

Another alternative is to replace an empty tank with a full tank but not replace thrusters, valves, regulators, etc. This requires a quick-disconnect-type fluid connection which has a leak potential. In the event this approach is chosen for the refillable tank, redundancy in some critical components, such as dual-seat disconnects and latching solenoid valves, may be desirable to provide the reliability for the system that is specified. For the requirement of zero g expulsion with liquid propellants, this approach may be considered superior to the alternative discussed below, because it provides for an automatic replacement of a positive expulsion device which may be cycle limited.

On-orbit refueling with no routine hardware replacement is a third option that has a number of advantages. The most obvious one is the cost savings, not only in reduced hardware requirements but also in making no demands on the satellite design to facilitate exchanging modules to satisfy the total impulse requirement. In its simplest concept, only a quick disconnect would be required to refuel the tank. This would be the least costly and lightest of the above concepts for a given propellant. The disadvantages are possible leakage at the disconnect and multiple-cycle demands on an expulsion device.

Finally, there is the situation where, either because of design constraints, refueling-in-orbit hazards, or the high cost of complete multi-propulsion modules, large enough tanks will be provided to contain all the fluids required for the mission. In such a case, the choice of the propulsion subsystem will be based on the previously discussed total impulse criteria. However, even in this case, the problem of fluid disconnects will face the designer.

In the modularization of propulsion subsystems, two principles will be involved: one is to include in a module all of the low reliability components (Category I) which may need refurbishment, and the second is to provide isolation valves so that a malfunctioning module could be isolated and the remaining modules carry on the attitude control functions into the next refurbishment cycle.

The FleetSatCom reaction control subsystem offers an excellent example of modularization, it is relatively complex, and it incorporates most of the features and component types that are used on satellite payloads. Figure F-4 is a block diagram of the present FleetSatCom reaction control subsystem. The blocks are arranged in the form of a system schematic such that the propellant flow path from the propellant tanks to the thruster nozzles can be easily traced. The system incorporates protection against likely single-point failure modes by the use of standby redundant thruster systems.

Figure F-5 illustrates (in block-diagram form) the application of modularity to the FleetSatCom type of reaction control propulsion subsystem. The entire Figure F-5 system would be a complete, self-contained propulsion module. The modules would be strategically located on the spacecraft to provide the required reaction control forces. Redundancy requirements would be met by simply adding additional standby modules. This approach has the advantage of ease of maintenance, since a complete system can be removed and replaced as a unit. The general approach also leads to the possibility of standardized propulsion system modules that may be used on a numb rent payloads.

This idealized approach could only be considered for Space Shuttle launches, since there are very large penalties to be paid in weight, volume, mass properties, and performance. Also, the block diagram has been simplified so that it does not include instrumentation, heaters, electrical control to valves, test ports, and structural interfaces.

Figure F-5 also illustrates a lower level of subsystem modularity. For instance, the use of interchangeable thruster modules has been demonstrated to be practical on several programs and should be emphasized as a standardization objective for the Shuttle era. Further modularization of other subsystem elements such as tankage and the gas

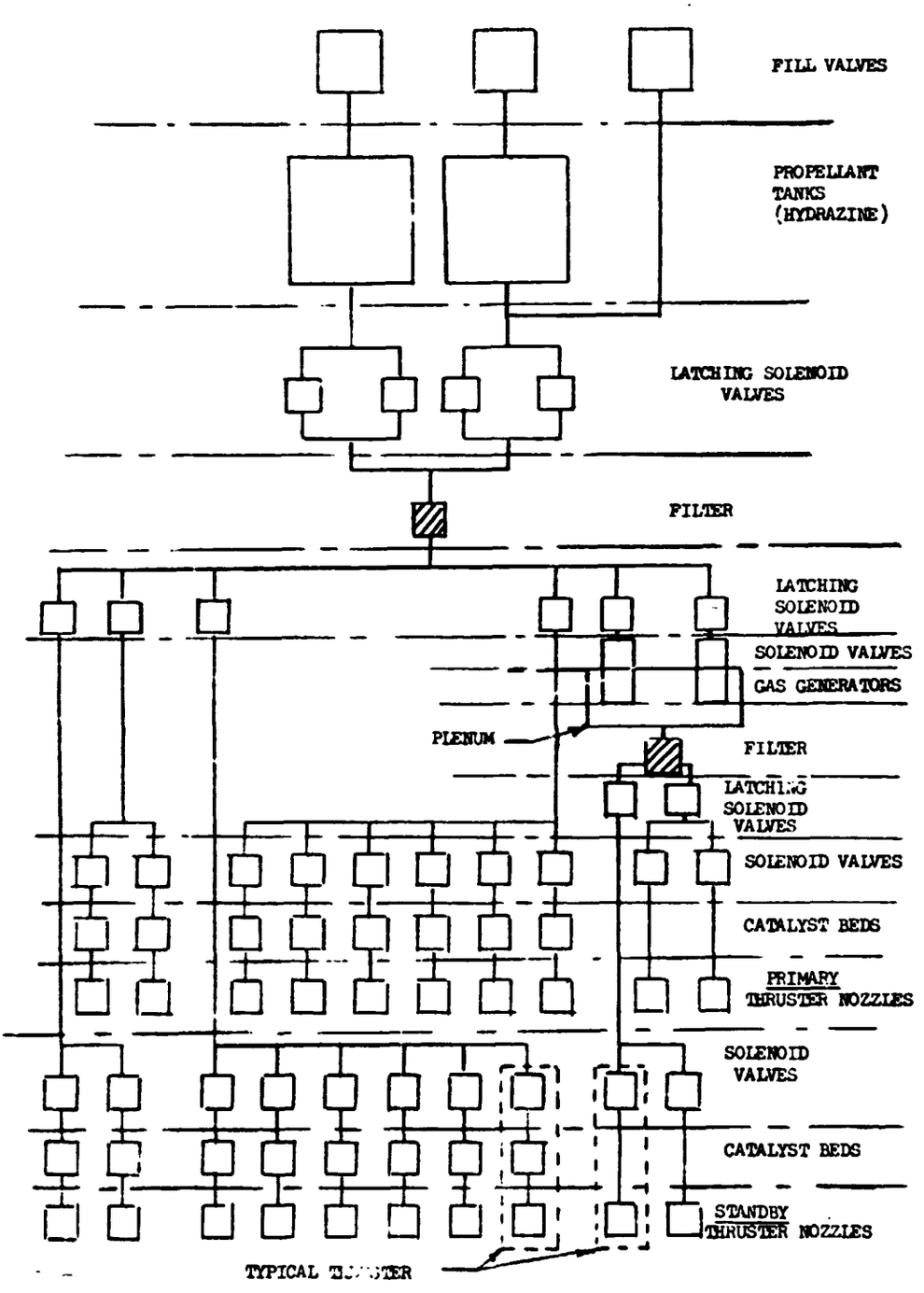


Figure F-4. Block Diagram of a Typical Satellite Propulsion Subsystem (FleetSatCom)

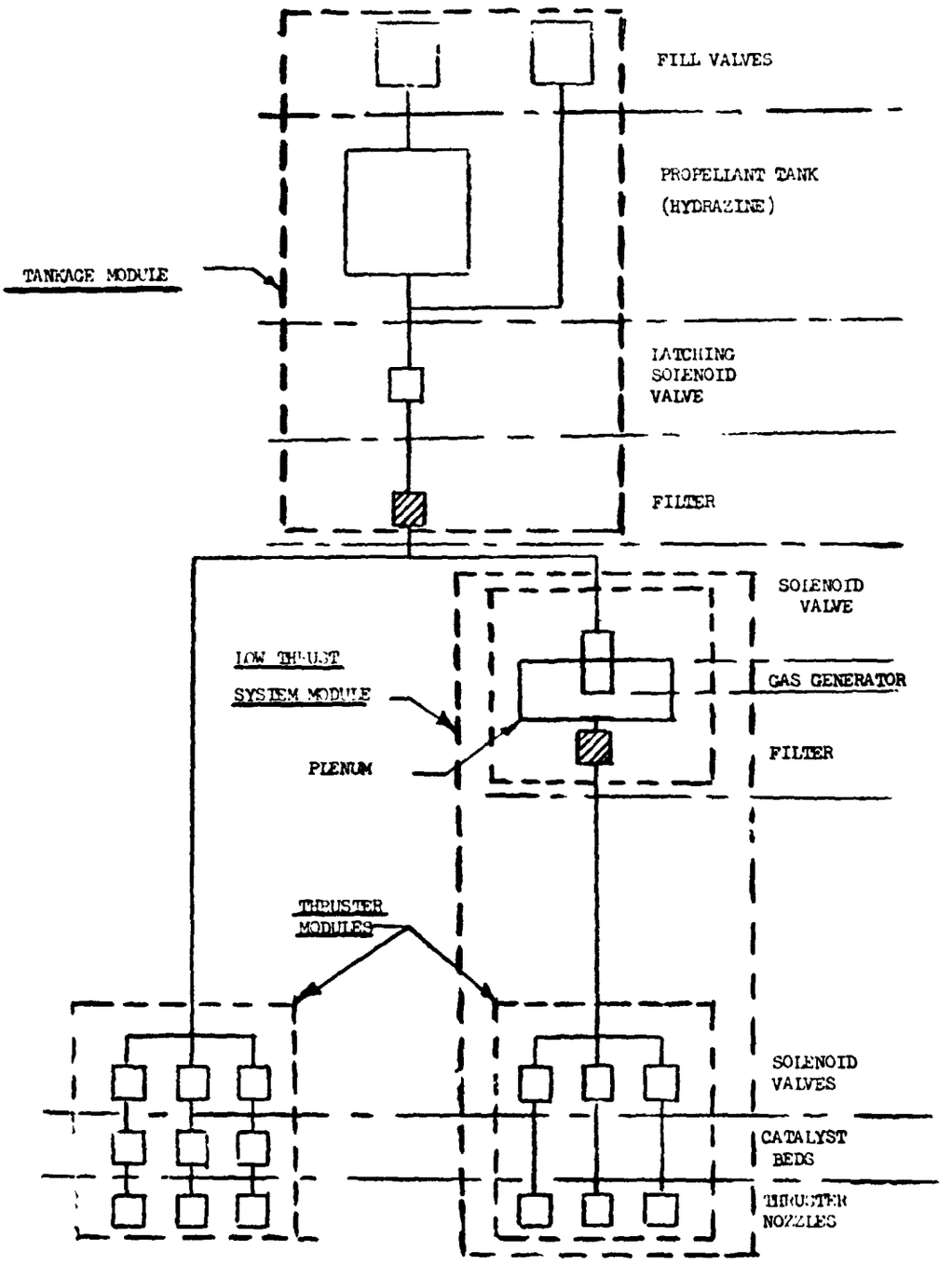


Figure F-5. Block Diagram of Modularized Propulsion System (FleetSatCom)

generator-plenum subsystem leads to a building-block approach which could provide cost savings through standardization. This approach is currently employed to a certain extent on Air Force programs, as each new program tries to make maximum use of previously qualified and flown hardware. This building-block approach of assembling standardized subsystem and subassembly modules into a propulsion system is generally the most effective approach, but it necessitates breaking propellant feed system attachments for module replacements.

If the propulsion subsystem has, for instance, a pressure regulator, the designer's option is to either include it in one of the replaceable modules or to provide a redundant regulator(s).

The question of redundancy in the attitude control subsystem has to be carefully considered because the docking maneuver will, in general, require an attitude-stabilized payload. A single failure or even a dual failure mode has to be considered either by module redundancy or by a backup cold gas system.

The problem of fluid disconnects was mentioned before. Drip-free fluid disconnects are at present available and a redundant seat arrangement, perhaps combined with an isolation valve actuated before a disconnect operation, could possible ensure hazard- and leak-free operation. Similar attention should be paid to electrical connectors which will probably lead to a departure from the standard multi-pin concept.

To take full advantage of the Shuttle era, consideration should be given to standardization of propulsion modules into two or perhaps three thrust levels to suit the variety of payloads anticipated by NASA with a resulting reduction in RDT&E and production costs.

Considering the refurbishment potential, the designer should weigh the possibility of relaxing some of the specifications for components such as thrusters (number of cycles), valve leakage criteria, and higher

safety factors which would reduce the cost of production and testing of the components. Similarly, the possibility of pre-fueled fluid tanks hermetically sealed should be considered which would eliminate prelaunch testing of the thrusters. Such a procedure is adopted at present in an ICBM system, and the bipropellant thrusters are only sample tested at the factory.

An important factor for the in-orbit replacement is the need for telemetry signals either to the Shuttle or to the ground which will give an indication of a module's health. Tank pressure and temperature, thruster pressure, propellant valve motion indicators and temperatures, and others will be monitored throughout the mission duration.

For all of the above, the Shuttle environment and crew safety criteria must be accommodated. This may mean venting partially expended modules being returned from orbit, providing means for propellant dumping of loaded modules and/or tanks, and orienting tanks with bladders to avoid subjecting the bladders to damaging loads during Shuttle ascent.

3. GROUND REFURBISHABLE

In the ground-refurbishable concept, the selection of an attitude control propulsion subsystem will follow the criteria of total impulse discussed before. The modular concept for a propulsion system will still be advantageous here, since on return to the ground, new or refurbished modules can be easily replaced, resulting in short turnaround time.

Since return of the complete payload to the ground can be considered as a more costly and complex operation than in-orbit refurbishment, more attention should be given to the degree of redundancy and component reliability for maintenance of attitude stabilization during the mission and docking maneuver with the required degree of reliability. Since no need for in-space refueling or module replacement exists, there is no problem with fluid or electrical disconnects.

The procedure during ground refurbishment in principle should be similar to the maintenance practices of scheduled airlines, and a study of airline procedures should be of value in establishing design requirements.

The telemetry requirements for monitoring propulsion parameters will be similar to those for on-orbit maintainable payloads.

The Shuttle concept provides the possibility of detailed examination of failed and/or matured components which will be an important factor in the hardware updating and evolution. This will result in longer life components, thus reducing future payload costs.

D. CONCLUSIONS AND RECOMMENDATIONS

When cost is the primary criterion, selection of reaction control subsystems of greater weight and volume and lower impulse efficiency may be preferred. This is possible with the Shuttle because of its capability for larger and heavier payloads. The propulsion subsystem selection is dependent on the mission; i.e., expendable, on-orbit maintainable, or ground refurbishable. For on-orbit maintainable systems (even for high total impulse missions), cold gas attitude control subsystems may be desirable. Tradeoff studies are required for selection of a cost-effective reaction control subsystem for each of the three payloads. Modularization of the RCS should lead to the standardization of two or three thruster levels for a variety of payloads.

APPENDIX G. ELECTRICAL POWER

Analyses should be performed concerning possible cost savings in the use of secondary batteries with solar cell arrays in Shuttle payloads. It is recommended that Shuttle payloads consider the use of short-life NiCd, AgZn, and Pb-acid battery designs for low earth orbit and inclined elliptical orbit spacecraft with replacement of the batteries at the end of their design lifetimes. Also, battery operating temperatures should be considered, since battery operating temperatures influence the cost of batteries. This cost must be compared to cost in the thermal control area from relaxed thermal design requirements to determine if overall cost savings accrue from wider battery temperature ranges. The analyses hinge upon the use of short-life battery designs for periodic battery replacement via the Shuttle.

The use of NiCd batteries for periodic replacement may be less expensive over a 10-year operational period in space than one designed to endure the entire mission duration. If batteries are designed for long life, they are typically designed for low depths of discharge. Conversely, a short-lived battery can be designed for deep discharge with little excess capacity. Since battery cost is approximately proportional to capacity, it is conceivable that smaller batteries replaced periodically would incur less overall cost than one set of large batteries without replacement. It can be seen from Figure G-1 that designing for shorter life with periodic replacement in synchronous orbit will always cost more than designing for 10 years. This is explained by the relatively few charge-discharge cycles in synchronous orbit.

The inclined elliptical and low earth orbit cases, however, show that a significant cost reduction can be obtained by periodic replacement of short-lived batteries. In these cases, the depths of discharge for the five-year design period are small (10-20 percent). Substantial gains in battery

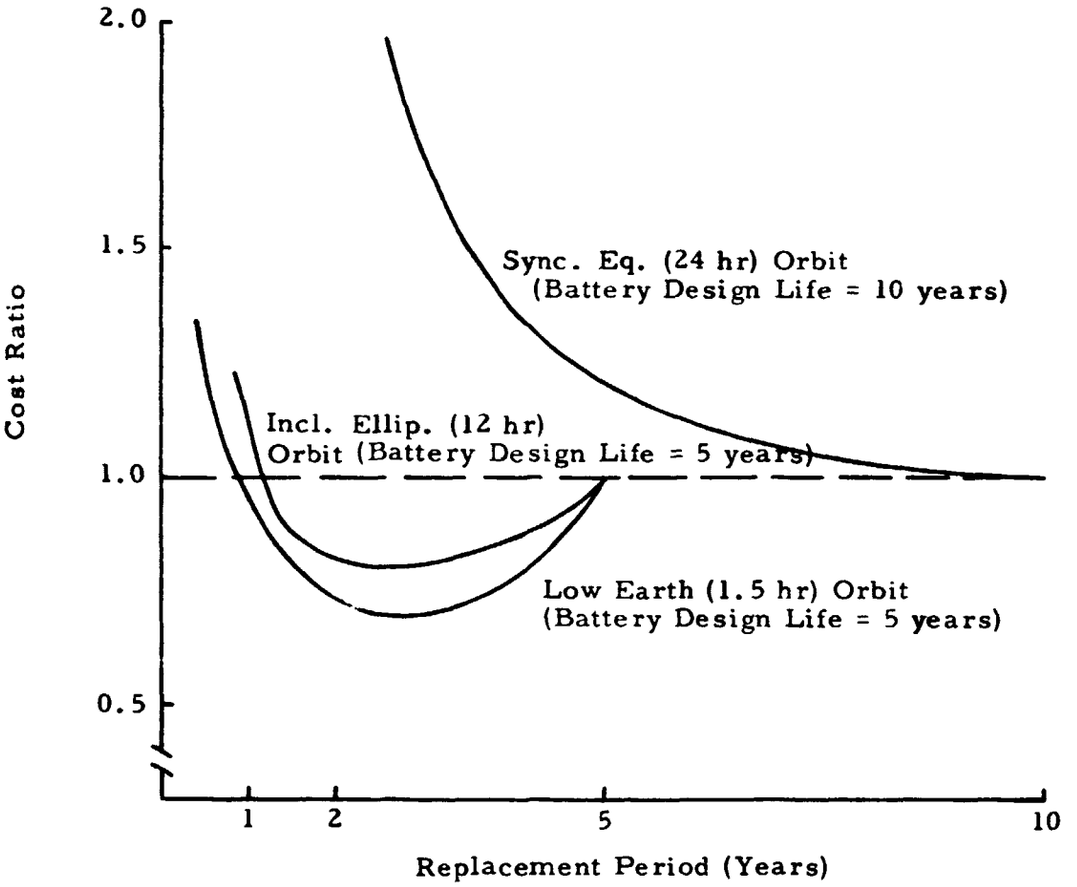


Figure G-1. NiCd Battery Utilization Results

utilization through high depths of discharge are realized for shorter lifetimes. Figure G-1 shows an optimum design life of two to three years. This would involve three battery replacements over a 10-year operational period and would save 20-30 percent of battery costs in comparison to replacing the batteries after five years.

A battery costs about $220 per kg ($100/lb). If launch costs were $880 per kg ($400/lb), the total cost of batteries for 10 years at $1,100 per kg ($500/lb) would be about $250,000 for a 500-watt spacecraft in either inclined elliptical or low earth orbit. Reducing this amount by 20-30 percent would yield savings of $50,000-$75,000 over a 10-year operational period.

Wide battery temperature ranges and particularly high battery temperatures are detrimental to battery life. Consequently, it is conventional to design for narrow battery temperature ranges in long-life applications. With the Shuttle, long battery lifetimes and narrow battery temperature ranges may not be necessary or desirable. Wider battery temperature ranges and lifetimes shorter than the mission life could be accommodated by periodic battery replacement.

Figure G-2 shows the effect of battery temperature on battery cost. Batteries normally experience excursions through some part of the 0-40° C range; for example, the batteries may be designed for a 0-10° C range. Thus the average cost incurred over a narrow range such as this must be evaluated and compared to the average cost over larger operating temperature ranges.

NiCd batteries are normally used in secondary (charge-discharge) battery applications because of their long-life capability. With periodic battery replacement, shorter-lived but less expensive battery types may be economically attractive such as AgZn and Pb-acid batteries.

Table G-1 summarizes the battery cost estimates where "i" represents the battery life (replacement period).

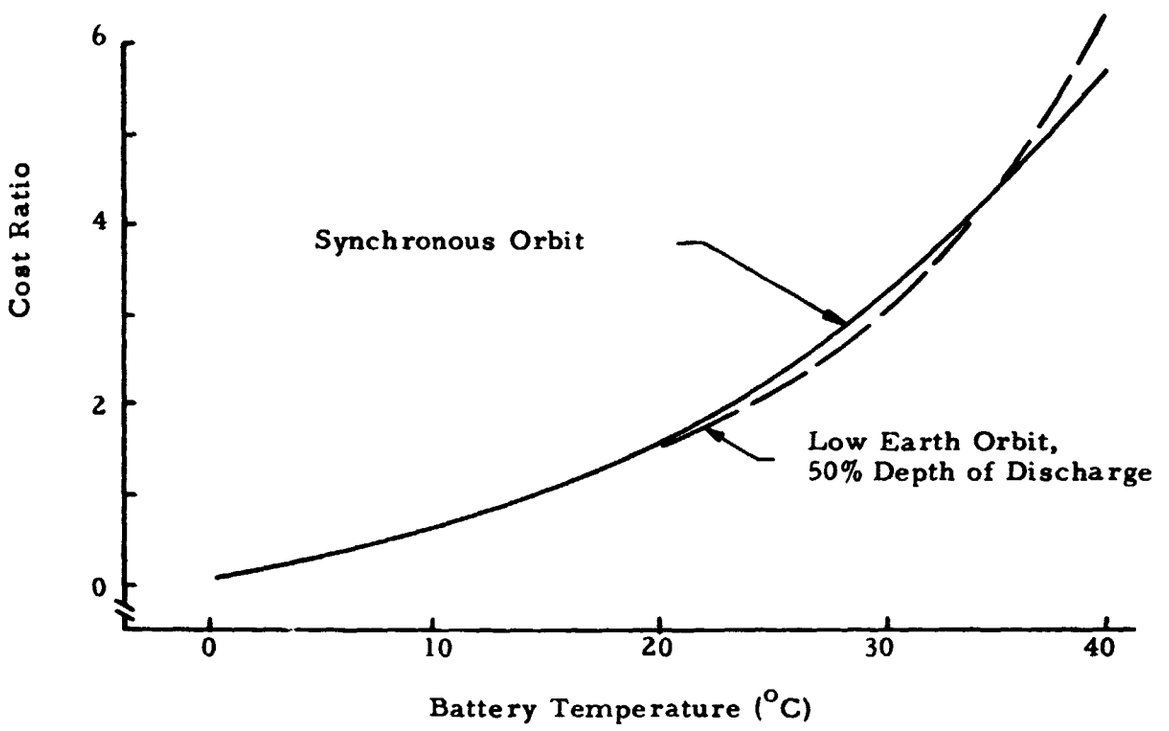


Figure G-2. Effect of Temperature on NiCd Battery Costs

Table G-1. Battery Costs Per Kilowatt

	Cost ($ / kw)	
	Low Earth Orbit	Synchronous Orbit
NiCd	100,000	20,000
AgZn	-------	$\frac{40,000}{i}$
Pb-Acid	$\frac{30,000}{i}$	$\frac{20,000}{i}$

For the AgZn battery in synchronous orbit, a life of several hundred cycles and several years seems likely. Sealed Pb-acid batteries show considerably greater cost savings potential. A lifetime of one to two years in low earth orbit would save 70-85 percent of the cost of NiCd batteries. Also, rapid charge techniques have been proposed which might allow greater depths of discharge and greater savings. In synchronous orbit where two or more years life seems likely, the cost saving relative to NiCd batteries would be substantial.

It should be noted that the magnitude of potential cost savings implied in the table above could be significantly affected by launch costs. If launch costs were assessed on a per kilogram basis, then low weight could be important to low total cost, and the situation portrayed in the table might not be realistic.

It is recommended that Shuttle payloads should consider short-life NiCd battery designs with periodic battery replacement for low earth and inclined elliptical orbit spacecraft as a means of significant cost savings.

Shorter-life battery designs may be economically attractive with periodic battery replacement. Also, long battery lifetimes and narrow battery temperature ranges may not be necessary or desirable. Wider battery temperature ranges and lifetimes shorter than the mission life could be accommodated by periodic battery replacement. The impact on battery costs of such a battery design approach must be assessed.

Less expensive battery types such as AgZn and sealed Pb-acid batteries may be economically attractive with periodic battery replacement.

The results are applicable to both on-orbit maintainable and ground-refurbishable Shuttle payloads. The batteries for expendable payloads by definition must be designed for mission duration. The Sortie payloads will use laboratory-supplied power.

APPENDIX H. ELECTRICAL DISTRIBUTION

This analysis is to provide an approach and a set of parametric data which will facilitate low cost payload design. Trade-off considerations are herein investigated for comparing centralized with decentralized power distribution methods. Figure H-1 illustrates the basic differences between the two concepts.

Because of the wide variety in the types and nature of the payloads, it became necessary to define some payload as a baseline for the comparison. The Defense Support Program (DSP) satellite was chosen as representative of the payloads to be carried aboard the Shuttle. The electrical load distribution was normalized to that of the DSP satellite. Other assumptions are as listed:

a. 28 volt raw power
b. 50 percent of satellite power to mission equipment
c. 50 percent of satellite power to vehicle electronics
d. Of the 50 percent for vehicle electronics:
 1. 53 percent for communications, data processing, and instrumentation
 2. 16 percent for attitude control
 3. 27 percent for stabilization and control
 4. 4 percent for communications and power control
e. All loads are assumed to require regulated power as a worst-case situation.
f. Cable losses are assumed to be insignificant for costing.
f. Loads range from 100 to 2,000 watts.

Figures H-2 and H-3 depict the decentralized and centralized power distribution modularization concepts, respectively. In each diagram the dotted blocks indicate the redundant partner where redundancy is required.

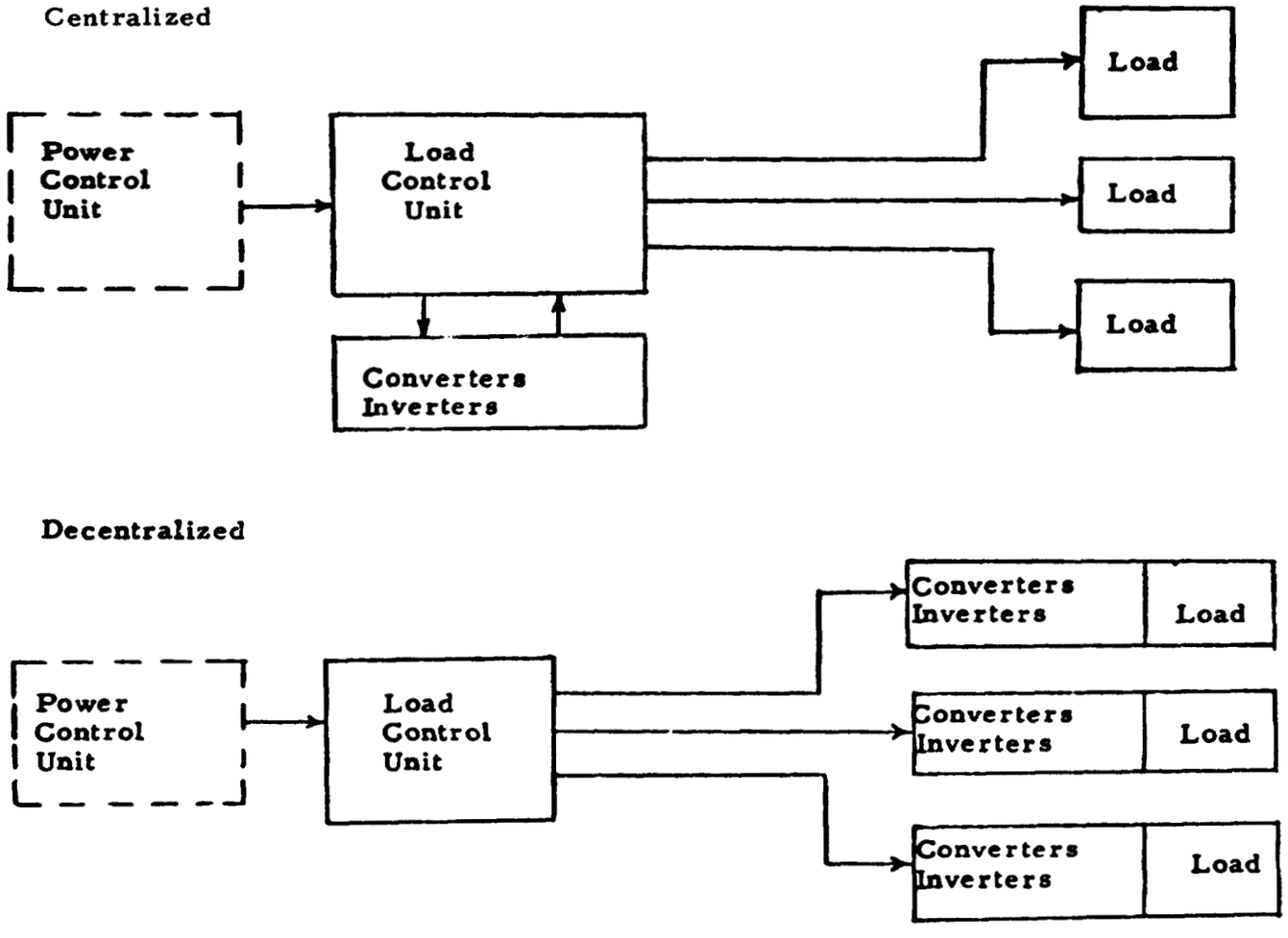
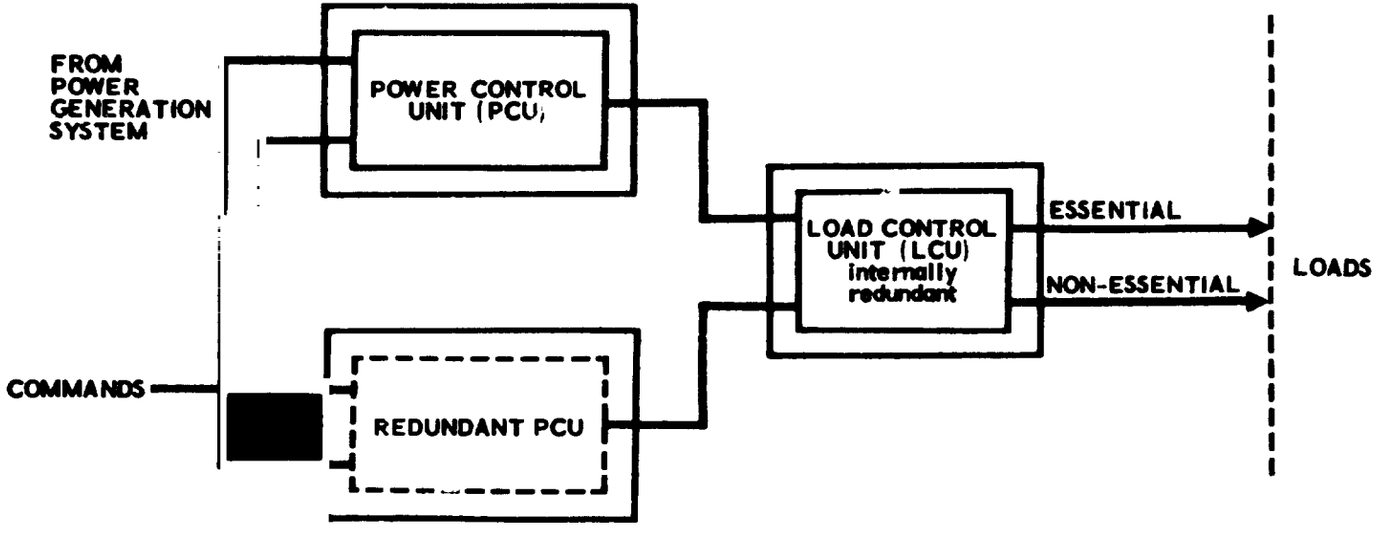


Figure H-1. Centralized and Decentralized Power Distribution System Concepts



CONTAINS: PCU -

- LINE REGULATOR
- PCU CONTROL
- BUSES

LCU -

- BUS & BATTERY CONTROL
- LOAD DISTRIBUTION
- BUSES

OUTPUTS:

- ESSENTIAL & NON-ESSENTIAL PRIMARY POWER

Figure H-2. Decentralized Power Distribution

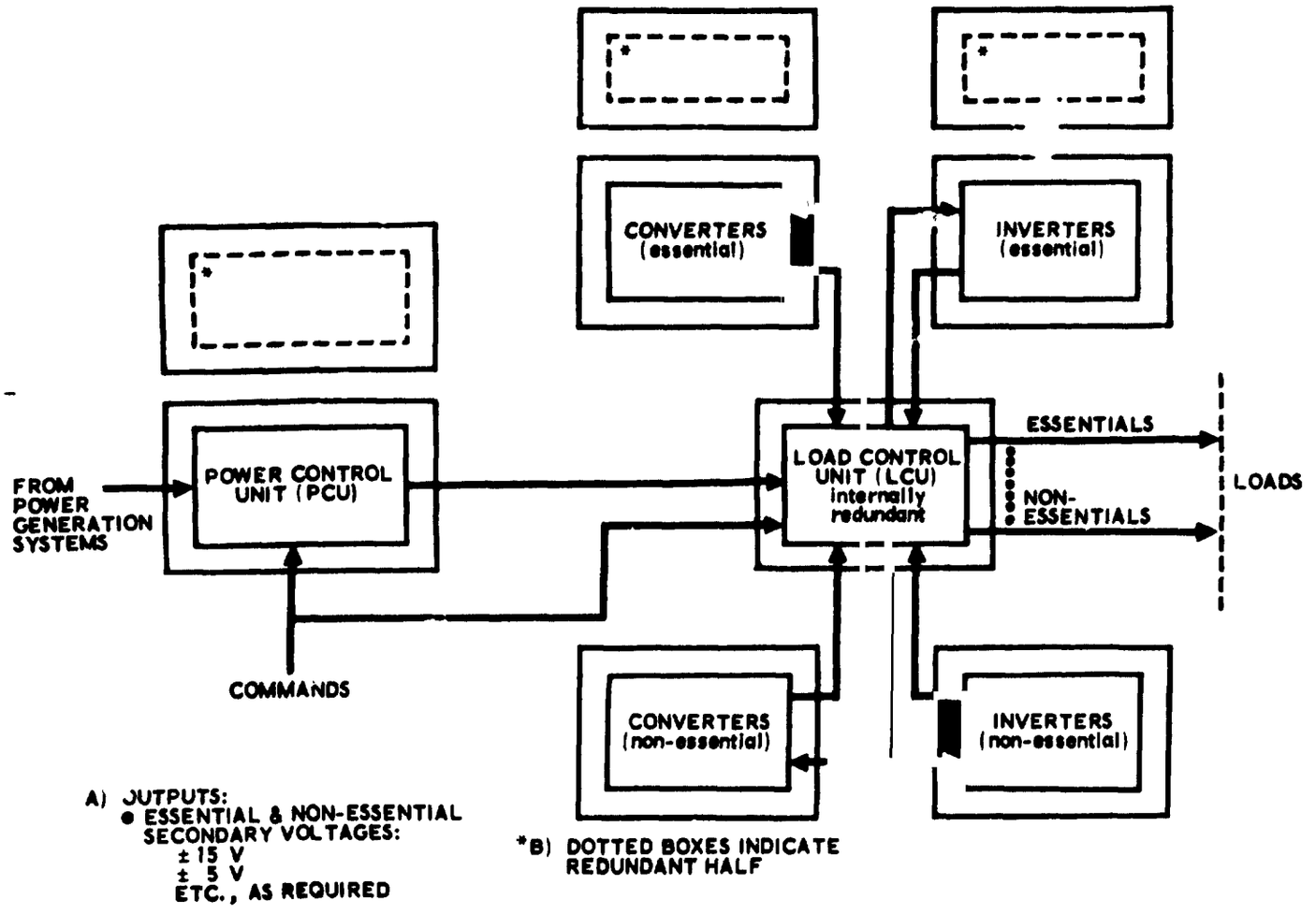


Figure H-3. Centralized Power Distribution

The heavy-lined blocks indicate how the subsystems can be modularized. It can be seen that the prevailing philosophy is that of configuring the modularization so that each redundant segment can be removed without disturbing the operation of the other redundant half. This prevents downtime of the satellite in the event that a unit has to be replaced.

In both concepts, raw spacecraft power is received by the Power Control Unit (PCU). In the decentralized system (Figure H-2) the PCU regulates and switches the primary 28-volt power to the Load Control Unit (LCU) which in turn routes this power to the proper user loads, essential or non-essential. The user loads will include localized converters which perform the function of converting the primary and 28-volt power to the secondary voltages required by these loads. The duties of the LCU for the decentralized system are much simpler than that for the centralized system, since only one primary power voltage need be distributed rather than many secondary voltages.

Referring to Figure H-3, the raw spacecraft power is received by the PCU, then processed and distributed to the LCU. All active power distribution functions (switching and selection of redundant loads or converters) and most fault-isolation operations will be handled by this unit. It receives the processed, regulated 28-volt power and routes it to the proper converters, receives regulated secondary voltages from the converters, and finally routes these to the proper user loads, essential or non-essential. Failure of an essential converter or load will cause the LCU to switch to an appropriate operable redundant unit.

The inverters supply a small percentage of the power; hence, cost impact as far as tradeoff analyses are concerned is minimum. Consider, therefore, the case of essential and non-essential converters. The weight/watt ratio decreases as the power output increases. Consequently, in a decentralized system with smaller converters, the weight will be greater than that of a larger converter in a centralized system. This is illustrated in Figure H-4. For instance, for 1000 watts power output and three mission equipments, the decentralized converters would weight about

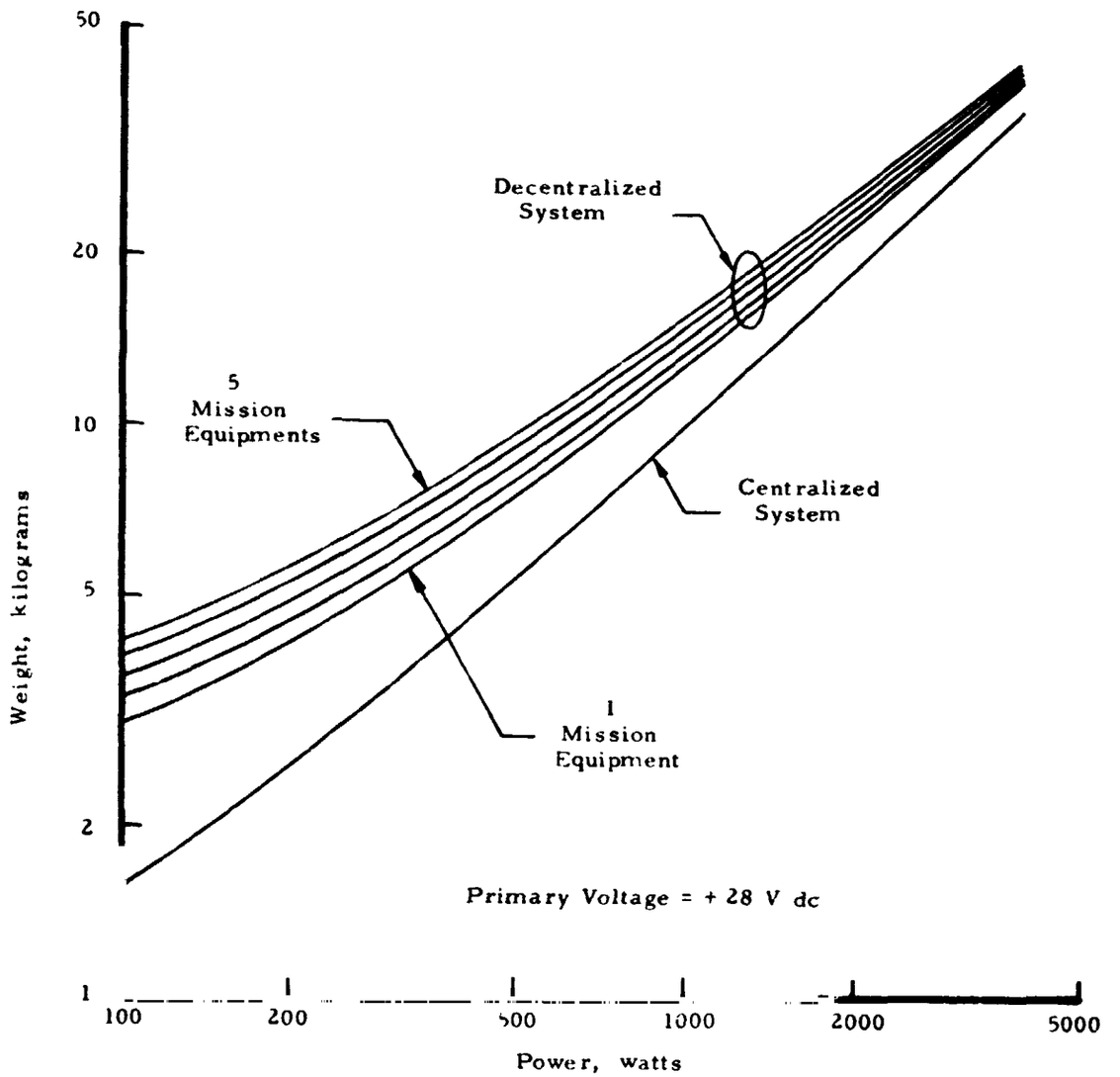


Figure H-4. Converter Systems Weight/Power Relationships

22.6 kg (50 lb), while the centralized converter would weight about 13.6 kg (30 lb). To this weight should be added the weight of the PCU and LCU. The centralized unit may weigh more because more secondary voltages must be distributed. However, it will be assumed here that the same weight applies for both cases of about 13.6 kg (30 lb) for the PCU plus LCU units.

The harness weight depends on the power transmitted, payload geometry, and level of redundancy. The wide variation in weights indicates the difficulties in assessing the cost. In any case, harness costs are expected to be a small part of the total power system cost and are therefore ignored in this analysis.

In assessing the cost figures for an electrical power system (less batteries and solar array), data are available which show that the first unit cost is directly proportional to weight. Thus a centralized system which is lighter than the decentralized one (by 9 kg for the 1000-watt case) will probably cost less. However, a centralized converter system with a single converter for all loads will imply a greater initial development effort, since small converters for a decentralized system of proven design are readily available. The development cost of a new system is an order of magnitude higher than the first unit cost. Consequently, it will be less costly to use a decentralized system, particularly since the small weight penalty involved will probably be acceptable for Shuttle payloads.

There are also other factors in favor of a decentralized system. A major drawback of a centralized system is that of reduced flexibility. Major changes in payload power requirements could lead to an expensive redesign effort. In a decentralized system, the change would only affect the particular module supplying the load, and another converter could most likely be found among existing designs. A summary of other factors favoring the choice of a decentralized power supply are tabulated in Table H-1.

On the basis of this evaluation, a decentralized power conditioning system is a candidate configuration for a lower cost effect. A decentralized system offers lower overall weight and complexity, reduces

Table H-2. Power Distribution Comparisons

PARAMETER	CENTRAL POWER CONDITIONING	DECENTRAL POWER CONDITIONING
Power Distribution	More complex, requiring switching and control for as many secondary voltages as may exist, plus heavier buses, increased cable complexity	Less complex; most (unmanned) vehicles currently employ decentral power
Bus Protection	Each load requiring multiple power voltage inputs must have a bus protection device for each voltage. The number of circuit breakers increases for each power system. Solid state switches and circuit breakers must be designed to operate at each bus voltage.	One circuit breaker per load. Solid state switches and circuit breaker designs required for only one voltage.
Power Conversion	High power levels at low voltages and tight regulation requirements will result in high development costs. New development item. Cable weights high due to the high currents at low voltages; Filters and regulators are required in digital equipment.	Transformer rectifier designs are well proven. Power converter designs exist.
Isolation	Isolation between electronic equipments difficult to accomplish due to the sharing of common power supplies and power distribution systems.	Isolation provided by power converter in electronic equipments.
Flexibility	Changes to power quantity, quality or voltages levels can force a power converter redesign late in the development program. System has a limited capability to accommodate changes	Growth potential is good. Changes to electronic equipment that affect power conversion are accomplished within the equipment
28 VDC Power Requirements	28 VDC power is still required for non-electronic equipment (valves, solenoids).	
Losses	Distribution system losses are higher due to distributing power at low voltages.	
Reliability	Reliability is reduced due to increased system complexity.	
Off-the-Shelf Hardware	Redesign required to remove power conversion hardware and repackage.	Some hardware may be used directly.

cost, increases reliability, and lends itself to modularization of decentralized converters to allow individual replacement for fault isolation.

These conclusions apply equally to the on-orbit maintainable, ground-refurbishable, and expendable types of payloads. Since the Sortie payload is retained with the Shuttle during the mission duration, its power is received from the Shuttle itself. No separate power source in the payload is necessary. Load distribution equipment for special Sortie experiments may be required in the payload.

APPENDIX I. BACKUP SYSTEM FOR RETRIEVAL

A. INTRODUCTION

Study has been performed to determine a method of providing sufficient control for retrieval assuming primary vehicle attitude control is lost. The study examined various modes of primary attitude control system failure and the consequences of these failures on payload dynamics. Since docking is a prerequisite for on-orbit maintainable and ground-refurbishable payloads, a system must be implemented to assure that retrieval can be accomplished in the event of a payload failure.

In order to perform this analysis on a backup system for retrieval, the Large Space Telescope (LST), as proposed by NASA/MSFC (Ref. I-1), was selected for analysis. This payload design includes control moment gyros (CMGs) for fine attitude control and cold gas reaction control system (RCS) thrusters for acquisition, powered flight orbit adjustment, initial CMG spin up, and emergency operation. The CMG momentum management is provided by electromagnetic torque control. Since this vehicle represents most of the possible failure modes of three-axis controlled vehicles, it is believed that the analyses conducted apply to this general class of payloads. The analyses are described in the following paragraphs.

Typical primary control system malfunctions were assumed. These included:

1. Electrical power failure during the CMG-controlled mode; could include a total vehicle power failure
2. Hardover CMG torquer; could be caused by a failure in the torquer positioning circuit; e.g., an open feedback transducer
3. Roll hardover command during RCS-controlled mode; could be caused by an attitude sensor failure (The roll axis was chosen because, for this vehicle, roll attitude acceleration is the highest.)

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4. Single RCS thruster control valve stuck open; could be a mechanical failure; i.e., the thruster would remain on although commanded to off

The resultant effects on vehicle motion, possible fault detection schemes based on these effects, possible corrective actions, required backup control equipment, and backup equipment available in the LST were determined for each of the assumed primary control systems malfunctions.

Assuming that no backup control system was available, the steady-state spin rate about the principal axis of maximum moment of inertia was also determined for each malfunction.

Assuming that the steady-state uncontrolled spin was attained, and a backup system was available, the impulse required to stabilize the vehicle and maintain control was determined for each malfunction.

Based on results of the above analyses, three operational alternatives were studied:

1. Docking with the uncontrolled satellite after it attained steady-state spin
2. Switching the primary system off and the backup system on upon malfunction detection
3. Switching the primary system off upon malfunction detection and switching the backup system on after the satellite had attained the steady-state spin, possible shortly before rescue vehicle launch

B. ASSUMPTIONS

The LST vehicle characteristics are described in Reference I-1. Figure I-1 shows a block diagram of the attitude stabilization and control system, and Table I-1 summarizes the system hardware. Nomenclature therein is as follows:

1. Wide-angle Sun Sensors (WASS) - Provide inertial attitude reference for transfer to the coarse and fine pointing modes

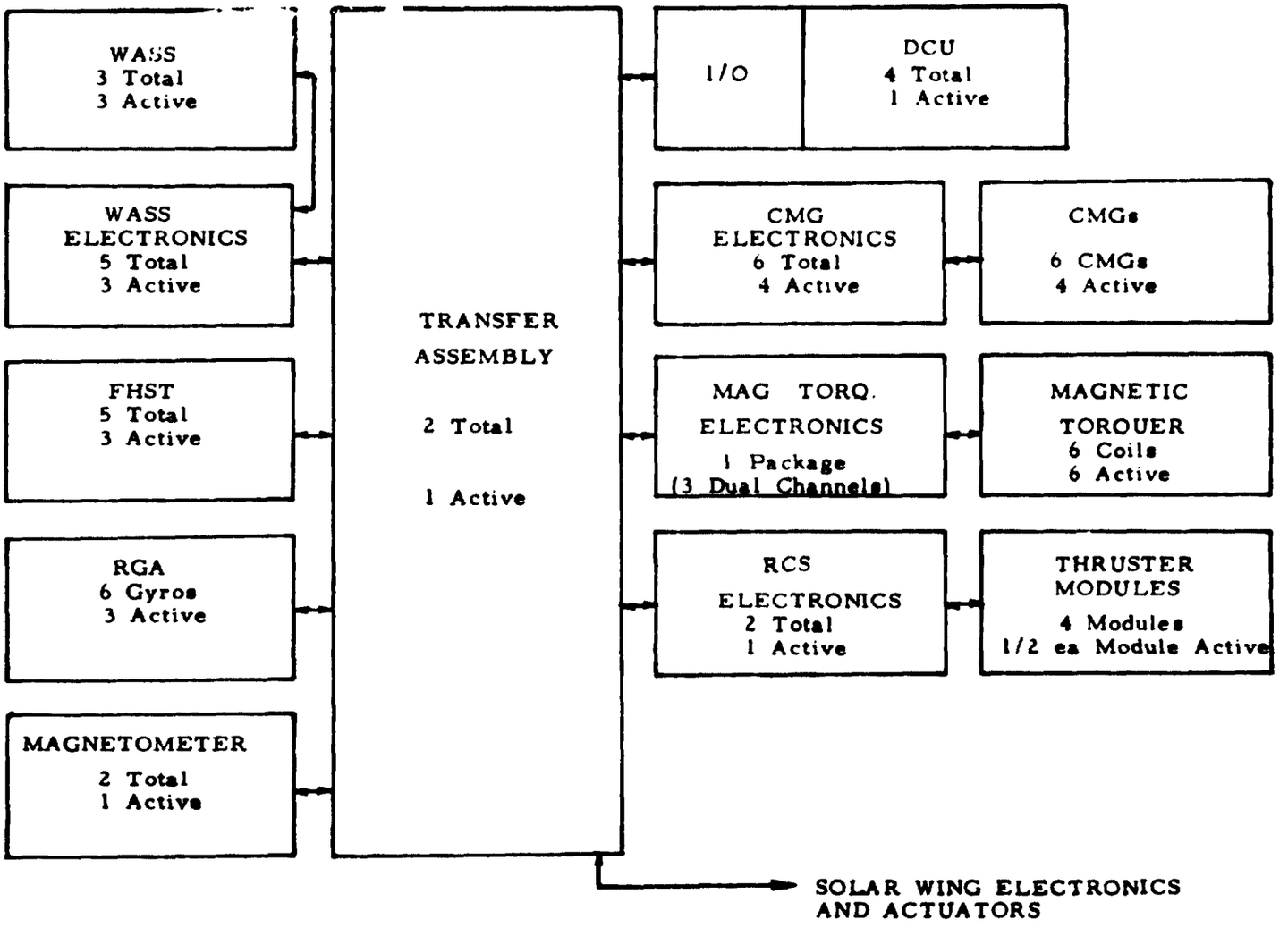


Figure I-1. LST Attitude Sensing and Control Systems Block Diagram

Table I-1. LST Attitude Sensing and Control System Hardware Summary

Item	Number of Units	Unit Weight kg (lb)	Average Unit Power (w)	Total Weight kg (lb)	Total Average Power (w)
Wide Angle Sun Sensor	5	0.45 (1)	0	2.3 (5)	0
WASS Electronics	5	0.45 (1)	0.4	2.3 (5)	1 (3 active)
Fixed Head Star Tracker (Electronics Included)	5	6.8 (15)	14	34.0 (75)	42 (3 active)
Gyro Package (2 Gyro)	3	5.0 (11)	15	15.0 (33)	45
Magnetometer (3 Axis)	2	2.9 (6.5)	2	5.9 (13)	2 (1 active)
Control Moment Gyro (339 newton-meter-sec)	6	61.2 (135)	(32 spinup)	367 (810)	36 (4 active)
CMG Electronics	6	12.2 (27)	18	73.5 (162)	72 (4 active)
Magnetic Torquer	6	14.1 (31)	6 peak	84.4 (186)	30 (peak)
Magnetic Torquer Electronics (3 Dual Channels)	1	2.3 (5)	5	2.3 (5)	5
Digital Computer	4	9.1 (20)	66	36.3 (80)	65 (1 active)
Transfer Assembly	2	4.5 (10)	3	9.1 (20)	3
RCS Electronics	2	7.3 (16)	11	14.5 (32)	11
RCS (All GN_2 on LST)				358 (790)	41

2. Fixed-Head Star Trackers (FHST) - Provide inertial attitude reference for reference alignment and coarse pointing mode

3. Reference Gyro Assembly (RGA) - Provides rate and position information in all control modes, provides the only attitude reference during star tracker or guide star occultation periods and slew maneuvers

4. Magnetometers - Measure earth's magnetic field for electromagnetic control computations

5. Digital Control Units (DCUs) - Receive data inputs from the various sensors and provide outputs to the CMG, magnetic torquer, and RCS electronics; also provide gyro compensation using other sensor and/or ground update information

6. Transfer Assembly (TA) - Provides interface to bring configurations of sensors and actuators on-line as required, routes signals, and provides failure isolation; also provides low level processing for fault detection, signal processing, switching logic, and command implementation and includes emergency operational circuitry to permit the spacecraft to acquire the sun using only this circuitry in conjunction with the WASS, PGA, and RCS

7. Control Moment Gyros (CMGs) - Provide control torques for vehicle stabilization in the coarse and fine CMG control modes. [Six CMGs are oriented (as shown in Figure I-2) so that their momentum vectors are constrained to lie in the planes of the six sides of a right hexagonal pyramid, of which the slope of the sides of the pyramid is 524 mrad (30 deg) with respect to the base.]

Four active CMGs provide momentum values up to 678 N-m-sec (500 ft-lb-sec) in the X-axis, 1017 (750) in the Y-axis, and 1186 (875) in the Z-axis. Only three active gyros are required for satisfactory ASCS performance; therefore, four failures are required before performance is degraded. However, this high degree of redundance may not be available in other vehicles. Therefore, the results of reduced control were examined briefly, as will be shown later.

The reaction control system is as shown in Figure I-3. It is assumed that signals from the pressure transducer just ahead of the

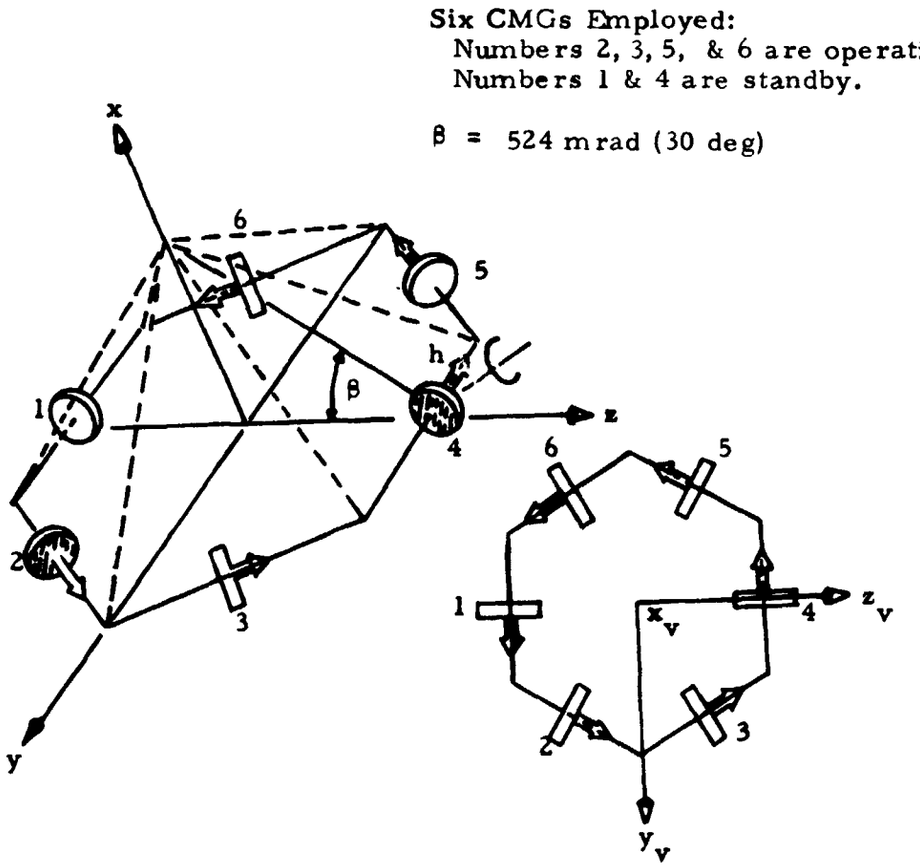


Figure I-2. Orientation of CMGs on LST

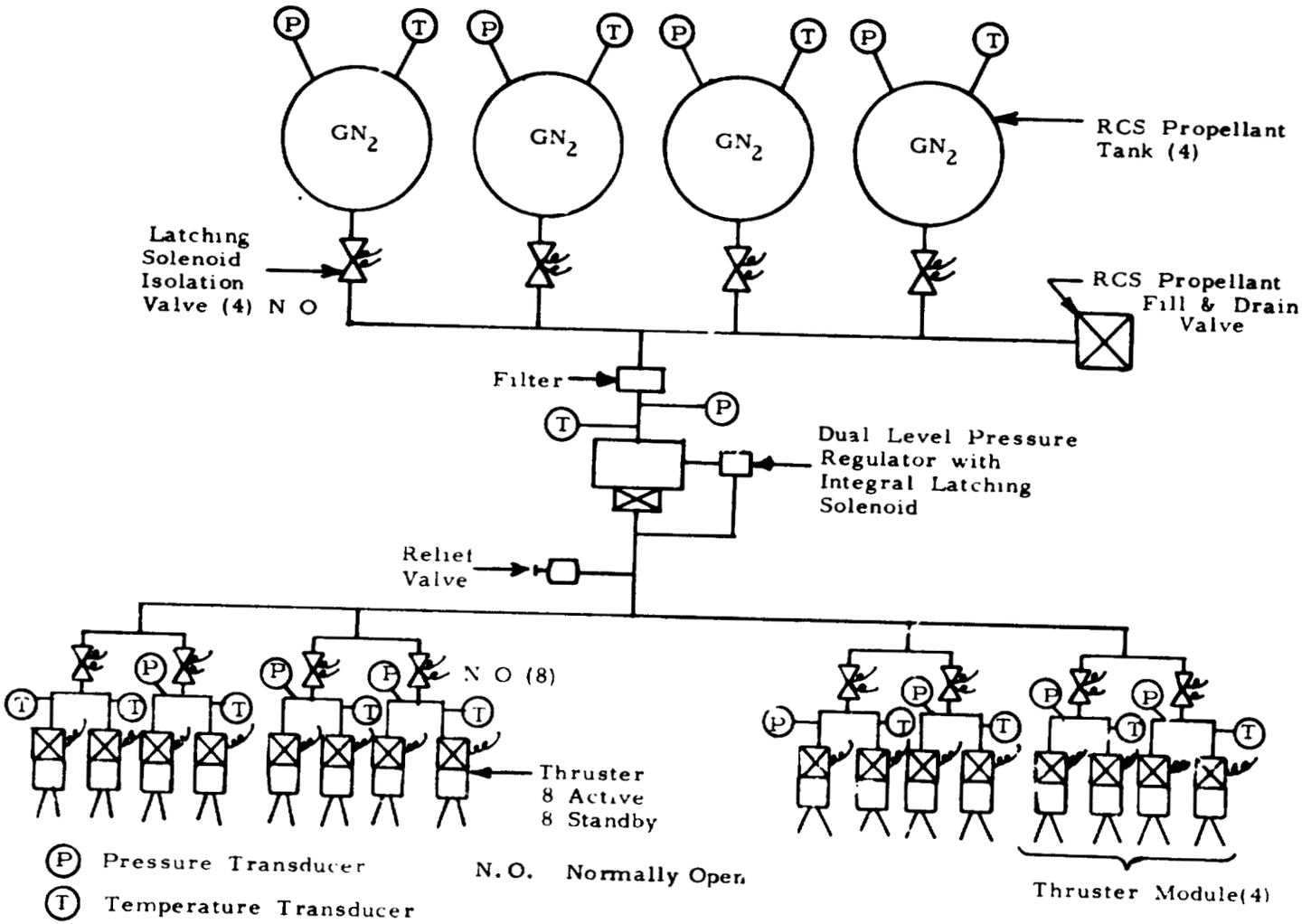


Figure I-3. System Schematic of the Baseline LST RCS

pressure regulator will be used to switch open the GN_2 tank isolation valves in sequence as the gas is used. Each of the four thruster modules contains two active and two standby thrusters. Each pair of thrusters can be isolated by the normally open latching solenoid valves shown; therefore, should a thruster stay open, it is assumed that the pair would be shut off by applying power to the closing solenoid of the isolation valve. The four GN_2 tanks have a total capacity of 114 kg (252 lb) of gas with a specific impulse of 70 seconds. The thruster modules are assumed to be installed as shown in Figure I-4 with the following characteristics:

	Pitch	Roll	Yaw
Thrust level per thruster ~ N (lb)	44.5 or 2.2 (10.0 or 0.5)	44.5 or 2.2 (10.0 or 0.5)	44.5 or 2.2 (10.5 or 0.5)
Minimum impulse bit per thruster ~ N-sec (lb-sec)	0.44 (0.10)	0.44 (0.10)	0.44 (0.10)
Moment arm ~ m (ft)	4.15 (13.60)	1.52 (5.00)	4.15 (13.60)
Vehicle moment of inertia ~ kg-m^2 (slug-ft^2)	125,000 (92,400)	17,700 (13,040)	130,000 (95,700)
Angular acceleration (10-lb thrust level) mrad/sec^2 (deg/sec^2)	2.910 (0.167)	7.660 (0.439)	2.840 (0.163)

C. ANALYSIS

1. ELECTRICAL POWER FAILURE WHILE SYSTEM IS IN THE CMG-CONTROLLED MODE

As stated earlier, it was assumed that 4 CMGs are active normally and that they can provide up to 1186 N-m-sec (875 ft-lb-sec) of momentum in the Z axis. If electrical power to the CMGs is interrupted, the rotor spin motors, as well as the gimbal torquers,

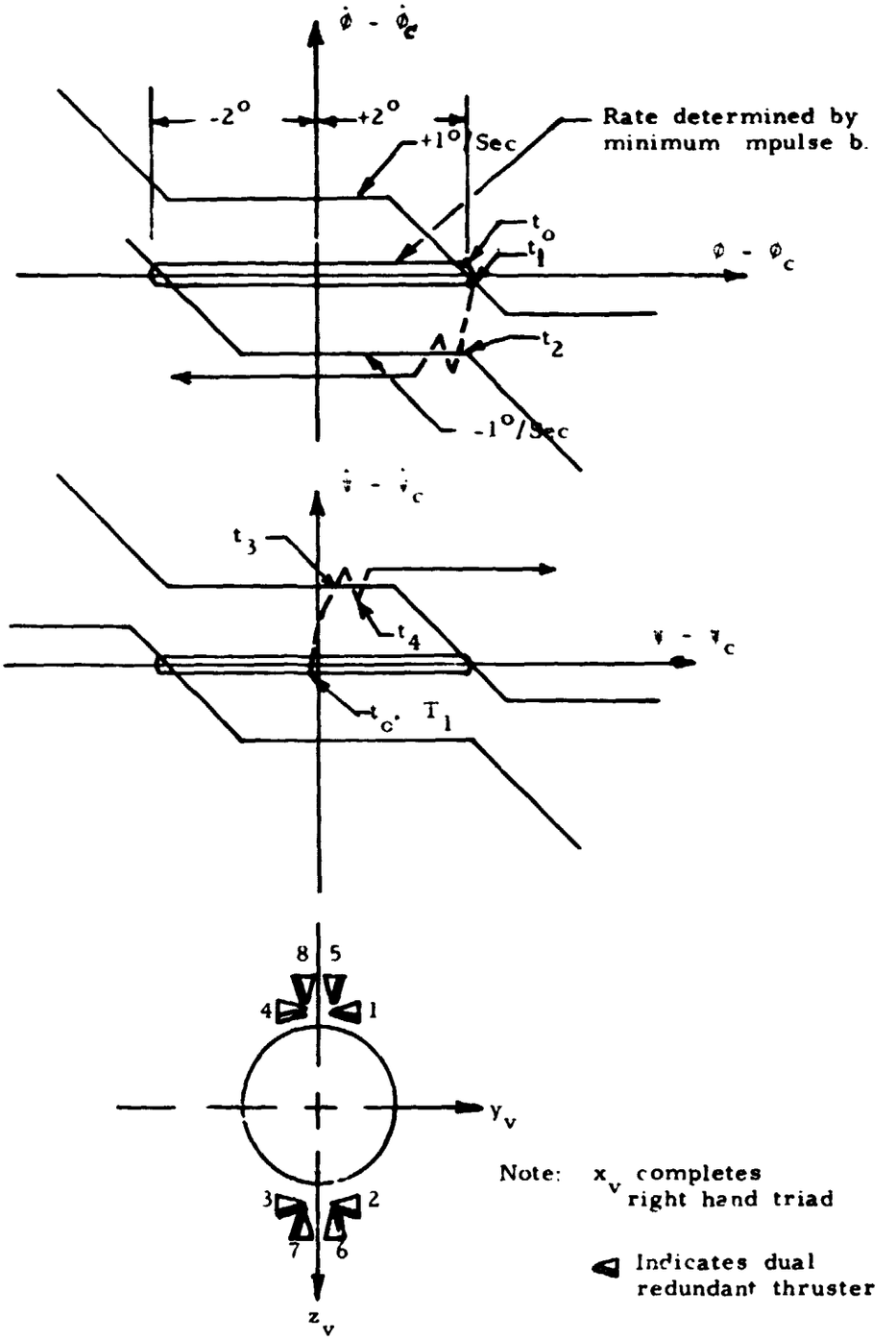


Figure I-4. Roll and Yaw Phase Plane Loci and Thruster Arrangement

de-energized. If no external disturbing torques are present, there will be no initial effect. If external torques are present, the system will have sluggish response; however, the vehicle will still remain under control, because the spinning rotors will oppose the change in momentum required by the external torques. Eventually, due to energy dissipation, the rotors will spin down, and their momentum will be transferred to the vehicle. In the steady state, assuming the absence of external torques, the vehicle will attain a constant spin rate about the principal axis of steady-state spin rate, as shown in Table I-2 was found to be about 9.08 mrad/sec (0.52 deg/sec). Table I-2 also shows that about 87.2 N-m-sec (64.3 ft-lb-sec) of RCS impulse or 0.42 kg (0.92 lb) of GN_2 would be required to stop the spin.

It can be seen that the pitch moment of inertia is very nearly equal to the jaw moment of inertia. Under these conditions, the spin about the yaw axis is lightly stable, so that any relatively small disturbing torque will produce rather large and slow-subsiding vehicle oscillations about the other two principal axes.

The above considerations are important with respect to the feasibility and advisability of docking with the uncontrolled vehicle. More about this will be discussed in the following Section D.

The most obvious way to detect this failure is to monitor the rotor speed. Of course, a more general vehicle power failure would be indicated by a power-off electrical power system status signal. Either of these signals could be used to switch-in the backup RCS (or the primary RCS in the LST if it is operational). Whether the backup RCS is switched-in at this time or just prior to rescue vehicle launch is subject to other considerations which will be explained later.

Backup equipment required for this failure, as well as those included in the LST design are listed in Table I-2.

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Table I-2. NASA Shuttle Payload Backup Control Criteria

Malfunctions	Electrical Power Failure in CMG-Controlled Mode	Hardover CMG Torquer	Hardover Command in Roll in RCS-Controlled Mode	Single RCS Thruster Hangs Open
Effects	1. Initially none. 2. Increasing rates and attitude errors. 3. In steady state vehicle spins about I_{max}-axis.	1. Reduced control capability in CMG control mode. 2. Increasing rates and attitude errors if insufficient control capability left.	1. Increasing roll rate. 2. Increasing roll attitude error.	1. Roll and yaw LC[5] about max LC rate switching-level initially. Steady roll plus sinusoidal pitch and yaw later (if allowed) SS spin about I_{max} axis still later (if allowed).
Parameters Indicating Malfunctions	1. CMG rotor spin rates decrease. If failure more general, power-off status signal.	1. Gimbal position hardover indication. 2. Increasing rate and attitude error if insufficient control capability left.	1. Increasing roll rate. 2. Increasing attitude error. 3. Increasing rate of propellant usage. (Probably too slow to be useful)	1. Roll and yaw rates stay close to limits. 2. Roll and yaw attitude errors increase monotonically. 3. Hardover signal on thruster opposite malfunctioning one.
Corrective Action	1. Switch in backup RCS with ground command prior to rescue vehicle launch.	1. Switch off malfunctioned torquer power automatically. 2. Switch in backup RCS with ground command prior to rescue vehicle launch. (If insufficient control capability left.)	1. Switch off malfunctioned RCS system automatically. 2. Switch in backup RCS with ground command prior to rescue vehicle launch.	1. Switch off malfunctioned RCS system automatically. 2. Switch in backup RCS with ground command prior to rescue vehicle launch.
Backup Equipment Required	1. Complete backup RCS - Thrusters, RCS electronics, low accuracy rate gyros, propellant supply, lines and valves, electrical power supply. 2. Backup command receiver. 3. Switching logic.	1. None, if enough control capability for docking remains. 2. Complete backup RCS - Thrusters, RCS electronics, low accuracy rate gyros, propellant supply system, electrical power supply. 3. Backup command receiver. 4. Switching logic.	1. Complete backup RCS - thrusters, electronics, low accuracy rate gyros, propellant supply system, electrical power supply. 2. Backup command receiver. 3. Switching logic.	1. Complete backup RCS - thrusters, electronics, low accuracy rate gyros, propellant supply system, electrical power supply. 2. Backup command receiver. 3. Switching logic.
Backup Equipment Available in LST[1]	All except rate gyros, but includes redundant attitude gyros.	All except rate gyros, but includes redundant attitude gyros.	All except rate gyros, but includes redundant attitude gyros.	All except rate gyros, but includes redundant attitude gyros.
Steady State Uncontrolled Motion	θ_{ZSS} 9.03 mrad/sec $(a^2_r)_{max}$ = 0.0061 m/sec^2	$\theta_e = \Psi_e$ = 0.005 mrad[4] θ_c = 0.005 mrad[4] If enough control remains. Larger att error and rates otherwise.	4 tanks depleted: ω_{xi} = 6.75 rad/sec, $(a^2_x r)_{ap}$ 28.5 g's ω_{sss} = 922 mrad/sec 1 tank depleted: ω_{xi} 1.69 rad/sec $(a^2_x r)$ 1.8 g's ω_{sss} 240 mrad/sec[3]	$\dot{\theta}_i$ 17.5 mrad/sec $\dot{\psi}_i$ = 6.72 E-5 cos (0.015 t) mrad/sec $\dot{\phi}_i$ = 17.5 + 8.81 E-7 cos (0.015 t) mrad/sec ω_{sss} 2.97 mrad/sec
Impulse Required to Stop SS Spin[2]	286 newton-sec = 0.42 kg GN_2		19,600 newton-sec 28.6 kg GN_2[3]	1,260 newton-sec = 0.14 kg GN_2

(1) The LST backup emergency mode switches on the RCS, WASS, RGS, with logic in the transfer assembly. It maintains up to 30 days X-pop attitude. This is acceptable; however, is not optimum WRT impulse storage requirements.

(2) Impulse requirement for unforced limit cycle attitude control: $\theta_D = \phi_D = \Psi_D$ = 34.9 mrad
Min impulse bit ($\int \Delta t$) = 0.89 newton-sec
Propellant Usage Rate = 0.0045 kg GN_2/hr

(3) The impulse requirement for this case is based on a yaw hardover command. The corresponding ω_{sss} for one tank depletion would be 627 mrad/sec. Maximum acceleration at solar panels would be 0.039 g's.

(4) Corresponds to fine pointing mode.

(5) Limit Cycle.

2. HARDOVER CMG TORQUER

As shown in Table I-2, this failure would have negligible effect except for reduced control capability in this mode. It, in a particular vehicle design, insufficient control capability were left after this failure, possible corrective actions and backup equipment would be required. The Table also lists the backup equipment available in the LST design. With respect to the "Steady State Uncontrolled Motion" listed for this type of failure, it should be clarified that the attitude errors and rates actually pertain to controlled motion in this control mode since it is assumed that enough controll capability remains after the failure. Substantially reduced control would produce larger attitude and attitude rate errors.

3. HARDOVER COMMAND IN ROLL IN RCS-CONTROLLED MODE

This type of failure would create an ever-increasing roll rate until either the malfunctioning RCS were switched out or the propellant were depleted. If one full tank of propellant were allowed to deplete, Table I-2 shows that the resulting roll spin rate would be about 1.69 rad/sec (97 deg/sec), and the resulting centripetal acceleration at the tip of the solar panels would be approximately 1.8 g's. After a rather long period, depending on energy dissipation rate, this momentum would be transferred into the principal axis of maximum moment of inertia, producing a yaw rate of about 226 mrad/sec (13 deg/sec). The impulse required to stop the steady-state spin is also shown.

In case of a depletion of four tanks of GN_2, the resulting rotational rates and centripetal accelerations would be much more severe, as shown.

It is clear that this failure requires fast switch-off of the malfunctioning RCS. Whether the backup RCS is switched in immediately or delayed until just prior to rescue vehicle launch is subject to other considerations to be discussed later.

4. SINGLE RCS THRUSTER CONTROL VALVE STUCK OPEN

The sequence of events following this failure depends on complete system dynamics and design characteristics; however, it is believed that the results obtained in the following analysis illustrate the most severe case.

Figure I-4 shows typical phase-plane roll and yaw loci for off-on control. The pitch plot would be identical but is not needed for this discussion. Figure I-4 also shows the thruster arrangement. Simultaneous firing of 1 and 2 or 3 and 4 would produce yaw moments plus lateral acceleration in the Y-axis direction, and simultaneous firing of 5 and 8 or 6 and 7 would produce pitch control plus lateral acceleration in Z-axis direction. On the other hand, firing any roll control thruster alone would produce roll plus yaw in addition to some lateral acceleration in the Y-axis direction.

It is assumed that the following sequence of events develops:

a. At time t_0, the roll and yaw states are as shown, and thrusters 1 and 3 fire to correct the roll error.

b. At time t_1, the roll command signal is switched off, but thruster 1 remains open, so the negative roll rate keeps increasing until, at time t_2, it hits the -17.5 mrad/sec (-1 deg/sec) roll rate limit. Some yaw moment develops in this time interval, but, due to the low yar acceleration, no significant yaw motion occurs.

c. At time t_2, the autopilot commands thrusters 2 and 4 to fire in order to stop the roll. Thrusters 1, 2, and 4 are now firing. The increased yaw acceleration causes the yaw rate to increase, until, at t_3, it reaches +17.5 mrad/sec (+1 deg/sec) limit.

d. At time t_3, the autopilot commands thruster 3 to open in order to stop the yaw rate. Thrusters 1, 2, 3, and 4 are now open.

e. At time t_4, the autopilot commands thrusters 2 and 3 to close because they are opposing and cancelling each other. At this time, the magnitude of the roll rate is something less than -17.5 mrad/sec (-1 deg/sec), and the yaw rate is approximately +17.5 mrad/sec (+1 deg/sec). No net moment is applied since only thrusters 1 and 4 are firing.

There are many possible sequences following time t_4, but it is felt that this is the condition of highest possible rates in pitch and yaw. It is assumed that, at propellant depletion time, this is the prevailing state; therefore, it may be assumed that the initial condition on the ensuing free body motion is: $\theta_o = 0.$, $\dot{\Phi}_o = -17.5$ mrad/sec (-1 deg/sec), and $\dot{\Psi}_o = +17.5$ mrad/sec (1 deg/sec).

D. RESULTS AND INTERPRETATION

Although for some failures, the resulting steady-state uncontrolled motion is a rather slow spin, it has been pointed out previously that this vehicle has two principal moments of inertia that are nearly equal. This characteristic causes the spin about the Z-axis to be very lightly stable so that, in the presence of external disturbing torques, rather large and slow-subsiding oscillations about the other two axes would be expected. Also, for the roll hardover command malfunction, the resulting yaw steady-state spin is shown to be at least 227 mrad/sec (13 deg/sec). This value is substantially larger than the allowable 1.75 mrad/sec (0.1 deg/sec) specified in the Orbiter docking specification (Ref. I-1)*. Furthermore, the acceleration at the solar panel tips due to the steady roll spin was found to be about 1.8 g's. This value exceeds the allowable 1.5 g's specified in Reference I-1.

These considerations indicate that, for three axis-controlled vehicles, docking with the body in an uncontrolled steady-state spin is not feasible for the roll hardover command failure and that prompt attitude stabilization is required from a backup system to prevent vehicle damage.

Docking with a tumbling body has been investigated (Ref. I-2). The preferred approach is to include a self-propelled, self-stabilized "third" vehicle designated as the "Module for Automatic Dock and Detumble (MADD)". However, until the feasibility and cost effectiveness of such a device are approved, backup control is preferred for three-axis active controlled vehicles.

*The feasibility of docking with a port normal to the Z-axis was being considered.

The above reasoning suggests that, at least for some malfunctions; e.g., roll hardover command in the RCS-control mode, automatic malfunction detection and primary control system switch out logic are required. It remains to be decided whether the backup system should be switched in immediately* and automatically or by ground command prior to rescue vehicle launch. Some considerations pertaining to these alternatives follow.

The amount of gas impulse required to restabilize the hardover command case would depend on the time required to sense the malfunction and on whether the restabilization occurs while the vehicle is still spinning about the roll axis or whether sufficient time has elapsed for the momentum transfer to the yaw axis to occur. This is due to the fact that the yaw RCS lever arm is 4.15 m (13.6 ft), while the roll lever arm is only 1.5 m (5 ft).** From this point of view, it would be more advantageous to switch in the backup RCS just prior to rescue vehicle launch.

Table I-2 shows that the amount of GN_2 required to maintain unforced limit cycle attitude control is very small (only about 0.0045 kg/hr). Due to external disturbance torques (at this altitude mostly gravity gradient and magnetic), the actual amount could*** be higher; however, it would still be very small, so that NASA's proposed 30-day RCS control would not impose a serious weight penalty. Nonetheless, this consideration also favors delaying the backup RCS switch-in.

The reliability of the backup system, once it is switched in, would be obviously higher if it is switched in later so that the required

* For the LST, there is enough sensor redundancy to be able to switch in the backup RCS immediately; however, if the backup rate gyros were the only available sensors, a minimum time of about 10 minutes would have to be allowed for gyro spin up and warm up.

** Rotational impulse is equal to $T(\Delta t)(\ell)$, where $T\Delta t$ is total impulse (N-sec) and ℓ is the torque lever arm.

*** It is well known that a forced limit cycle may require less ACS propellant.

proper performance time is reduced. However, reliability would be enhanced by elimination of the backup receiver and command link as explained below.

An advantage of automatic switching in of the backup RCS upon malfunction detection is that it would be autonomous and, therefore, independent of the ground command link. A backup command receiver (as listed in Table I-2) would not be required.* Another possible advantage of automatic immediate switching in of the backup RCS could be that mission operations and/or experiments could still be conducted until the rescue vehicle arrives.

A possibly significant disadvantage of immediate backup RCS switching to continue operation for a rather long period before rescue is that it may require a complete backup electrical power supply system.

It is apparent that a clear cut choice between the alternatives of switching in the backup RCS immediately upon malfunction detection or waiting until just prior to rescue vehicle launch is not possible. For each satellite design, then, the designer would have to evaluate considerations as explained above. For the LST, unless continued RCS operation for the 30 days is a hard and fast requirement, it is felt that delayed switching is to be preferred. If, on the other hand, prompt rescue launches can be assured, automatic switching would be preferred.

E. CONCLUSIONS

The foregoing analyses and results leas to the following conclusions:

1. Docking with the uncontrolled payload is not feasible for the roll hardover malfunction and is not recommended for the case of other malfunctions for the vehicle studied because of marginal Z-axis stability.

* A reliable telecetering down link would be required anyway, since it would not be reasonable to risk a rescue mission without prior knowledge of distressed vehicle attitude control.

2. A simple backup RCS should be provided. It should include, at least, the following equipment:

Component	Weight* (kg)	(lb)
4 Thruster Modules	9.1	20
1 RCS Electronics	8.3	16
3 Rate Gyros	10.9	24
1 Switching Assembly	4.5	10
Propellant Supply (1 Tank, valves, lines)	56.7	125
Propellant (GN_2)	28.6	63
Command Receiver	15.4	34
Electrical Power Supply	317.5	700**
Total	450.0	992

The LST includes the necessary backup equipment, except for the rate gyros, but includes redundant attitude sensors to provide the same function.

3. A malfunctioning CMG control system may have no significant effect; therefore, not all malfunctions of this type of system require switching the system off and the backup system on.

4. For those malfunctions that require switching off the active system, the switching off should be done immediately upon malfunction detection.

5. The alternatives of switching on the backup RCS system immediately after switch off of the active malfunctioning system, or switching it on just prior to rescue vehicle launch, must be evaluated for each vehicle design.

F. RECOMMENDATIONS

It is recommended that, until and if successful and cost-effective docking with a tumbling body as proposed in Reference I-2 is demonstrated, provisions be incorporated in on-orbit maintainable and ground-refurbishable

* Weights are typical and are extrapolated from data in Reference I-1.

** Assuming a completely redundant electrical power supply system for continued operation of at least 30 days.

satellite designs for stabilizing the vehicle after failure of the primary stability control system. The backup attitude control system should include, at least, the equipment listed in the "Conclusions" subsection of this appendix.

It is desirable, as in the case for the LST, that redundant primary control capability be incorporated in the vehicle design for reasons of mission operational reliability. In such a case, no additional backup control equipment is required.

The decision as to whether the backup control system should be switched in shortly after malfunction detection and primary control system switch off or first prior to rescue vehicle launch should be made for each design. Some considerations pertinent to this decision are included in subsection D of this appendix.

It should be noted that the analyses, results, conclusions, and recommendations herein are believed to apply, in general, to three-axis active-controlled vehicles. No investigation of the basic problem has been conducted with respect to other types of vehicles; e.g., spin-stabilized and gravity-gradient stabilized satellites.

G. REFERENCES

I-1. Large Space Telescope (LST) Preliminary Study, Volume II, Technical Report. National Aeronautics and Space Administration, Washington, D. C. (25 February 1972).

I-2. M. H. Kaplan, "Despinning and Detumbling Satellites in Rescue Operations," Paper presented at the 23rd Congress of the International Astronautical Federation, Vienna, Austria,

I-3. R. F. Thompson, Space Shuttle Baseline Accommodations for Payloads, Report MSC-06900, NASA Johnson Space Center, Houston, Tex. (27 June 1972).

APPENDIX J. GROUND OPERATIONS TIME

A. INTRODUCTION

Based on the massive amount of documentation related to the overall concept of the low cost Shuttle system, the review of information directly related to payload operations, and the results of a specific examination in a previous study (Ref. J-2), it was apparent that the present way of conducting prelaunch operations is not in consonance with the Shuttle concept. Prelaunch operations presently require from as little as 8 days for a very simple payload to as much as 60 days for a very sophisticated payload with a nominal median of 20-25 days. The foregoing data were used in the previous study and led to the realization of the time impact on the Shuttle. This subsequently caused the investigation of the potential for reducing the time required for prelaunch operation. The preliminary results show that it is feasible and reasonable to expect that prelaunch time can be reduced to the order of 10-12 days, or approximately half of the present requirement. This time increment has been the basis for comparison and a potential goal for achievement (see Figure J-1).

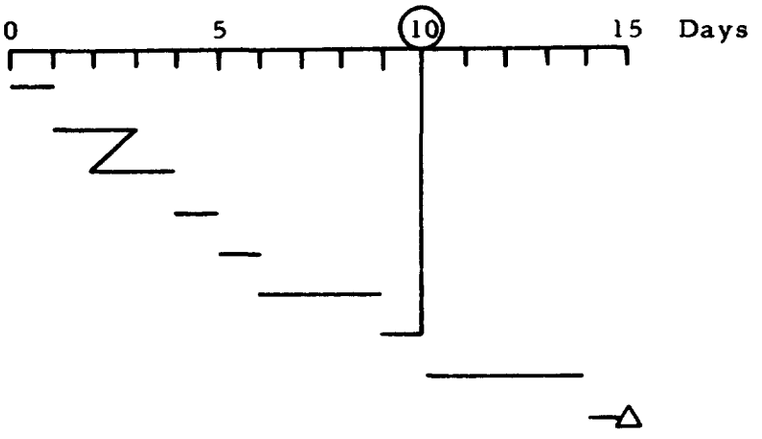
B. APPROACH

The approach was to analyze the prelaunch operations for a group of payloads representative of the four types of payloads considered; identify the prelaunch operations having the capability for reduction of time; consider changes for the design of the payload, the facility and equipment, and operational procedures; and develop candidate design guidelines for the accomplishment of time reduction for the payload prelaunch operations.

The prelaunch operations analyzed for potential time reduction were obtained for 24 payloads which are typical Shuttle payloads and which include examples representative of the four types specified for this study. The data sources were:

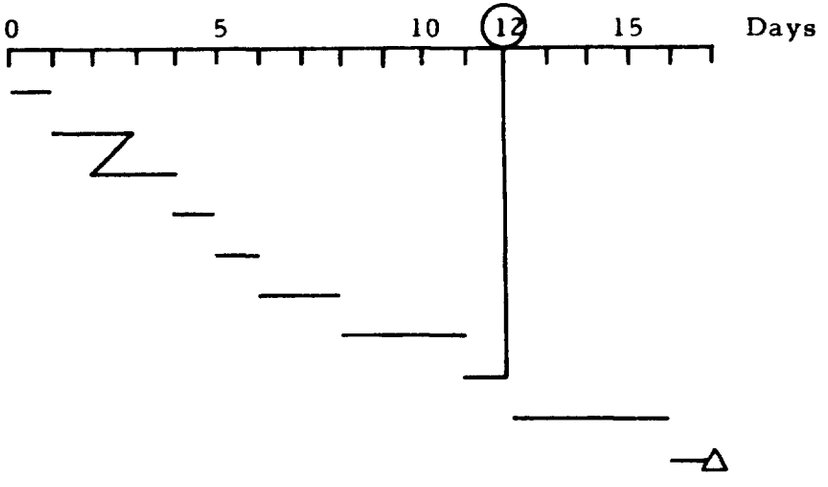
1. Martin Marietta Corp. Operations and Checkout Plan for Payload Carriers reports (Ref. J-3). Included 12 NASA payloads for the Shuttle.

SORTIE-LOW ORBIT



Arrival-Receiving and Inspection
Prep and Tests
Payload Functional Test
Pneu and Prop Test and Loading
Pyro Test and Loading and Final Preps
Orbiter Simulator
Orbiter Mate
Shuttle Operations
Count and Launch

HI-ORBIT-SORTIE



Arrival - Receiving and Inspection
Prep and Tests
Payload Functional Test
Pneu and Prop Test and Loading
Pyro Test and Loading
Tug Mate and Checkout
Orbiter Simulator
Orbiter Mate
Shuttle Operations
Count and Launch

Note: 1. Payload Arrives at Launch Base Ready for Launch
2. Payload Designed for Clean Interface

Figure J-1. Approach to Reduced Payload Prelaunch Operations

2. McDonnell Douglas Astronautics Co. DOD Space Transportation System Payload Interface Studies (Ref. J-4). Covered three specific DOD payloads for the Shuttle.

3. General Dynamics/Convair Aerospace. Space Tug Launch Site Service Interface Study (Ref. J-5). Covered typical payload/Tug operations.

4. Aerospace Corporation. Payload Analysis for Space Shuttle Applications Study (Ref. J-2). Included information related to a large number of predominantly NASA payloads.

5. Aerospace Corporation. Prelaunch Operations were obtained for five current payloads having typical and peculiar operations.

Using the data from the above sources, a matrix was developed incorporating a composite list of all of the prelaunch operations conducted on all of the payloads, and the time required for each operation was identified. These data were correlated and ranked in a list according to time and are presented in Table J-1. This list is not a complete list of operations, but it is comprised of those operations considered as having the potential for relatively significant time reduction and was used as a guide in selecting areas for further investigation.

An analysis of these operations was conducted to determine the source and reasons for the "excessive" time requirements, and efforts were concentrated on reducing the time needed for those items. The criteria used for the development of the design guidelines associated with these operations included consideration of completely eliminating the operation, modifying existing methods, combining and/or conducting operations concurrently, and developing new design for the payloads, the supporting equipment and the operations. In addition, full advantage was taken of the capabilities that may exist in the low cost Shuttle payload design philosophy (Ref. J-6), where the concept basically provides over-design and modular low density packaging. The assumptions made for this study are basically the same as those that were made for the preliminary approach to the 10-12 day timeline for prelaunch operations and consist of the following:

1. The payloads are complete and "flight ready" when received at the Prelaunch Processing Facility (PPF) on the launch base. Each unit is an all-up assembly, has been previously subjected

Table J-1. Time Ranking of Payload Operations

Operation	Work Shifts	
	Range	Median
Payload Functional Testing	1-14	7
Tug Mate	1-6	3
Preps for Mate to Orbiter and Tug	2-4	3
Interface Verifications	1-4	2
Leak Testing/Propellant Loading	1-3	3
Receiving and Inspection (1st Day)	1-3	3
Weight and Balance	1-2	2
Installations	1-2	1.5
Closeout Flight Preps	1-2	1.5
Simulated Flight	1-2	1
Ordnanc Checkout and Installations	1/2-2	1
Non-Hazardous Servicing	1	1
Contamination Control	1-2	1
Transportation and Handling (Each Instance)	1	1

to a full integrated system test (IST) and may require servicing. The assembly of certain small items which incur little time or risk may be excepted, such as protective covers/caps, special supports for transportation, and other accessory type items. It is recognized that a few payloads will require the assembly of some major items and these can be accommodated.

2. The payloads are generally "optimized" and modularized with respect to the standardization of the basic supporting structure and, to a large degree, the basic housekeeping subsystems.

3. The payloads are designed to utilize standardized test, checkout, servicing, and other support equipment which will be

furnished at the launch base for payload use. This specifically includes the Unified Test Equipment (UTE), the Launch Processing System (LPS), or other equivalent universal automatic type test equipment.

4. The payloads are designed to be man-safe. This includes fluid, propulsion, pyrotechnic, electric power, and all other subsystems.

5. The payloads are designed to retain a maximum amount of autonomy; interfaces with the Orbiter and Tug will be minimized.

While the foregoing assumptions are basically payload design guidelines, the realization of their incorporation into the payloads for the Shuttle is dependent on the detailed design guidelines as may be developed and implemented for each of them.

C. ANAL

1. RECEIVING, INSPECTION, AND INSTALLATION

The receiving and inspection of the payload have been shown to require as many as three work shifts, while 1.5 shifts are required for installing the various payload equipments. In order to reduce the total time for these operations, they will be conducted serially in the overall flow of operation and concurrently to the maximum extent possible. For this reason, the two operations have been treated together in this section.

It is expected that the time required for the combined operations can be reduced to not exceed two shifts because the payload will be delivered to the prelaunch processing facility in a "flight ready" condition, experienced launch base personnel will be performing many operations normally supported by payload personnel, and many of the operations will be conducted concurrently. During these operations, the payload will be removed from the transporter and installed on the payload processing fixture, and the management of the contamination control and associated equipment will be accomplished by launch base personnel. Concurrently with the receiving inspection of the payload, the transportation environmental monitoring record will be

examined, certain non-flight items will be removed, and the installation of the few payload equipment items will commence. The dedicated payload processing bay in the facility will be ready for the payload, the launch base support equipment will have been scheduled and available, and any payload-peculiar support equipment will have been integrated and the compatibility with the launch base system established prior to the arrival of the payload.

To accomplish this, two guidelines should be adopted:

a. The payload shall be basically designed for delivery in the "flight ready" condition, and the amount of equipment to be assembled following delivery shall be minimized to the greatest extent possible.

b. The design of the payload and any equipment to be assembled following delivery shall provide simple, fail-safe, low-risk interfacing hardware with sufficient accessibility to enhance the ease of assembly and preclude damage.

2. WEIGHT AND BALANCE

This operation typically requires two work shifts, or approximately 16 hours, during the payload prelaunch operations. The time is completely dedicated to the procedures; the concurrent performance of other tasks is precluded. The Space Shuttle Baseline Accommodations for Payloads document (Ref. J-7) generated by NASA requires that the precise payload mass center of gravity must be established prior to installation in the Orbiter; however, the requirements for maintaining the mass properties of the payloads far exceed the requirements for the purpose of determining the center of gravity of the loaded Orbiter. The weighing operation may be conducted prior to the prelaunch operations cycle. This is consistent with the receipt of a "flight ready" satellite, and it is recommended that the physical weight and balance operation be eliminated from the payload prelaunch operations.

With the elimination of this operation, the problem arises of controlling the mass properties of the payload in the event that module or other equipment changes become necessary during prelaunch or in-orbit operations. This may be handled by maintaining strict accuracy and control over the

payload data book and incorporating the following design guideline as appropriate to the particular payload:

> The interchangeability designed into the payload modules and other replaceable equipment will include the requirements for mass properties interchangeability.

The replacement of a payload module while on-orbit with a module that is heavier, due to updated design or other changes, also creates a payload mass properties problem. The following two guidelines provide an approach to solving this problem:

a. The payload design may incorporate ballast mounting receptacles, if necessary, in sufficient quantity and located appropriately on the payload to accommodate changes that may occur that violate the module interchangeability. The ballast may be added or removed manually as required on the ground. The design of the balancing system equipment will be required to have the capability for on-orbit changes to be performed remotely by the manipulator arms. Quick-disconnect tapered plugs of various materials may be considered as one approach for the weight/receptical design.

b. The payload design may incorporate electro-mechanical device(s), if necessary, to adjust the mass properties of the payload by moving ballast weights on command. The weights may be mounted on rods which are oriented properly to obtain the effective spatial weight location and may be moved by screw or other action. At least two types of such devices are presently being used successfully, one of which is used on a spinner and the other on a 3-axis controlled satellite.

3. LEAK TEST AND PROPELLANT LOADING

These operations are almost always conducted in series, and the time required for the overall procedure is nominally three work shifts. Presently, leak testing is accomplished by pressure/time monitoring and bubble or sniffer testing of each potential source, both of which allow to a limited degree the concurrent performance of other tasks but leave something to be desired in terms of accuracy. A third method used is to bag the entire payload, charge the system with an inert gas seeded with a low level

radioactive tracer gas, circulate the bagged mass of air through an analyzer, and monitor for total allowable system leakage. An accessory probe is used to isolate the individual source(s) if required. This method is effective and highly accurate but requires dedicated time for the bagged portion of the operation. Since this method readily lends itself to accomplishing a significant reduction in time, and is in consonance with verification testing of highly reliable payloads, it is recommended for use in the Shuttle. The payload, the launch base facilities, and the equipment associated with leak testing are similar to and are combined with those for the propellant loading operation.

The propellant loading and final pressurization operations are presently slow processes that are accomplished by a minimum number of personnel for safety reasons and require dedicated time. In addition, all other personnel not directly supporting the operation are evacuated from the entire area so that other tasks in the total support system as well as those associated with the payload cannot be performed concurrently. In order to reverse this situation and accomplish this operation in a safe, rapid manner, while allowing a large number of other tasks to be conducted concurrently, the following design guidelines should be considered:

a. Design the fluid systems for payloads and servicing equipment to working pressure-to-burst ratios sufficient to meet the safety requirements that will permit conducting other operations concurrently with fluid system servicing operations.

b. Design the fluid systems to minimize the number of sources for leaks.

c. Design to maximize the use of brazed or welded couplings for system components as well as the normal plumbing.

d. Eliminate the requirement for "one-more-time" flushing, draining, and purging during prelaunch operations This action may be taken by relying on previous operations, quality control, and by taking advantage of the system redundancy features and the elimination of single point failures incorporated in the payload design.

e. Increase the rate of fluid flow for loading and unloading operations. This may be accomplished by designing larger payload fill and drain lines, couplings, and larger capacity components associated with the systems.

f. The ground support equipment for both leak testing and fluid servicing should be designed as standardized equipment furnished for payload use at the launch base. The concept of this equipment is envisioned as consisting of self-sufficient, highly mobile units which will be used to service the several payloads in process in the prelaunch operations facility. The units may also be used for operations conducted at other on-base facilities including payload assembly and integration, storage, post-landing, and refurbishment areas. The equipment will be operated by experienced launch base personnel.

In addition to expediting propellant loading and unloading, the use of larger fill/drain couplings also provides capabilities that are both required and beneficial to the system. To provide the level of safety required for concurrent operations, the payload/ground system coupling design will incorporate multiple redundancy features to positively seal and lock the connections to assure the prevention of incipient or accidental leakage and unintentional disconnect. The safest design should be implemented, and decisions should not be made on the basis of payload vs ground equipment connector considerations. The larger coupling could also enhance the capability, safety, and time considerations in conducting on-orbit servicing operations, and the increased sizes of other components may be used to advantage in detanking or jettisoning operations.

By implementing the above design guidelines for the payload and the support equipment, the total time required for leak testing and propellant servicing should not exceed 1.5 work shifts.

Consideration should be given to factory (or launch site assembly) loading of propellants and gases with hermetically sealed tanks. No thruster testing is then possible in the prelaunch operation and propellant or gas valves can be only electrically tested for continuity of the circuit and valve stem motion (if proper indicators are provided). Such a procedure is adopted in one of the operational ICBMs and when applied here would result in simplification of the prelaunch procedure.

4. PAYLOAD FUNCTIONAL TESTING

Functional testing over the years has progressed toward the "ship and shoot" concept. This philosophy, while attractive, will probably never

be fully realized. However, further progress toward this end has been shown to be feasible and demonstrated to be possible and is in fact a necessary ingredient in the overall Shuttle concept. This has been reflected to varying degrees in all of the various studies and other literature related to the Shuttle in general and to payload operations in particular. The approach, as applied to satellite type payloads is not without precedent, since at least one program has successfully launched sophisticated satellites under the 10-12 day goal mentioned in this study and demonstrated the capability of further improvements. This is an exception and great effort in many areas was required in achieving this capability for one type payload. These same efforts will be necessary in implementing this approach for the Shuttle so that, by proper management of planning and design, the time required for testing of the large quantity of payloads to be processed may be reduced to contribute to the cost effectiveness of the overall system.

The time for functional testing has been shown to require as much as 14 work shifts; however, most of the payloads examined required times close to the median of 7 shifts. Approximately half of the payloads examined were based on their application to the Shuttle, and improvements in reducing the time required for testing were already reflected to some extent in the data. This could be seen in comparing the median of 7 shifts shown in this data to the 11.5 days shown for testing in the data collected for our previous study (Ref. J-2), which was based on the prevalent way of testing today.

In order to achieve the reduction of time to the level of approximately seven shifts, or three days for this study goal, the general philosophy previously mentioned must be implemented for the Shuttle payloads. The prime element in the recommended test philosophy is to reduce payload testing at the launch site, particularly during the operations in the prelaunch processing facility. The driving factor in reducing this testing is in lowering the level of the tests conducted. The extent to which the level of testing is lowered may be determined by tradeoffs, including payload reliability/availability

requirements, risks, time, costs, and other factors which will be different for each individual payload program. No attempt has been made to firmly define the level of testing. However, the trend and direction of the recommended philosophy is indicated and has been amplified sufficiently to show some of the methods for and the results of reducing the time for payload prelaunch testing to what is considered a nominal minimum limit.

In the planning of prelaunch test operations and in the design of the payload test equipment, and other support equipment, consideration should be given to the following guidelines:

a. In the design of the payload and the supporting test equipment, it should be emphasized that testing during the prelaunch operations will be to establish flight readiness verification. This is essentially status type testing, which is basically designed in most cases to isolate down to the replaceable module level. The guide to be used is that the prelaunch testing will, as a minimum, identically duplicate whatever on-orbit, prerelease, flight-readiness-type functional testing is defined for the particular payload.

 (Note: The capability will exist in the test equipment to conduct in-depth payload performance testing. However, this type of testing during the prelaunch operations cycle is not considered as the baseline, since it conflicts with the basic payload design and flight readiness verification testing concepts for the Shuttle. While such testing would require more time, and be expensive in terms of software and other factors for some programs, it may be implemented to accommodate justified cases. Generally, only critical portions of the payload should be subjected to this level of testing.)

b. Payloads should be designed to interface with and utilize universal-type test equipment. This equipment is characterized by the Unified Test Equipment (UTE) (Ref. J-8) or the Launch Processing System (LPS) (Ref. J-9) concepts, and may also be used in the payload ground refurbishment cycle and at the factory. This type of equipment has also been considered necessary (Ref. J-10) to accomplish the required 160-hour turnaround cycle (Ref. J-1) for the Orbiter.

c. Payload design shall include features that will contribute to the reduction of time required to perform configuration setup and teardown for test operations. These are presently extremely time-consuming operations and, while significant

reductions are inherent in the reduced testing concept, further reductions may be realized. Payloads should be modularized and incorporate simplified standardized test connectors of all types, builtin hat antennas, alignment aids, and other small items that would not require removal from the satellite. Location of payload equipment should consider concurrent operations and eliminate requirements to reorient or otherwise move the payload.

d. The payload design should provide whenever possible for automatic, remotely controlled, or fully accessible calibration features for the applicable equipment.

e. Payload equipments requiring highly precise physical alignment should be designed for the accomplishment by self-contained means wherever possible. Replaceable equipment will conform to interchangeability design requirements or have been previously assembled to the payload and designed alignment retention capabilities will be utilized. Manual alignments will be minimized to the maximum extent possible but where required will be accomplished using payload references. Accessory optical aids may be designed as an integral part of the payload or temporarily mounted to accurate payload reference positions.

f. Payload and support equipment design shall include provisions for module replacement in a safe rapid manner. Requirements for accessibility and orientation may be driven by ground rather than on-orbit operations due to the presence of folded antenna, solar arrays, support fixtures, test equipment, and a 1-g environment.

g. The design of payloads will include the incorporation of electrical interface test points that are an integral part of the interfacing equipment. Tests are delayed and large amounts of time are often expended in determining if an indicated problem is in the payload, the test equipment, or the various interfacing connections. Incorporation of these points will reduce the time required to isolate malfunctions and identify the offending unit without breaking the interface, provide a simplified rapid means to positively verify an interface, and may be used in all phases of ground testing.

5. TRANSPORTATION AND HANDLING

The ground transportation of the payload, and the associated handling and preparations, are shown to typically require one full work shift each time the payload is moved from one place of operations to another at

the launch base. Some of the movements that have been proposed in normal flows of operations include moving to a safe area for propellant loading and pressurization, to a special area for EMI tests, to other areas for simulator and Tug mating, and still others for simulator and Orbiter mating. Movements in some of the more specialized cases include to and from facilities used for payload sterilization, installation of specialized pyrotechnics, and short-term and long-term storage.

In addition to the above prelaunch transportation operations, there are those associated with the on-base payload refurbishment cycle operations, the movements to and from the launch stand in accomplishing the quick-change of payloads, and the transportation of payloads from the factory to the launch base. While the design guidelines are directed toward reducing the time for prelaunch operations, they also apply to the other areas and may be used in the support of all four types of payloads.

The design of the payloads should include as an integral part of the payload standardized equipment designed for supporting, hoisting, and otherwise handling the payload and which will interface and be used with standardized handling equipment. The general criteria for payload equipment design would include:

a. Simplified equipment that will directly interface with the ground equipment without the requirement to assemble/disassemble special adapters.

b. Multiple use capability to allow the use of horizontal support fixtures having the capability of re-orienting the payload in pitch and roll and for Orbiter manipulator arm attachment.

c. Compatibility with standardized hoisting slings for mating the payload horizontally and the handling gear used for mating the payload vertically.

The payload should be designed, the flow of prelaunch operations should be planned, and the facilities and equipment provided to minimize the number of times the payload is required to be moved. Among other advantages, this will reduce the potential for payload damage, eliminate requirements for retesting, enhance the control and maintenance of contamination,

reduce the amount of support facilities and equipment required, and result in a significant reduction in the time for prelaunch operations. The criteria for reducing these moves include:

d. The payload will include the design required for safety sufficient to permit all operations to be conducted in the payload processing facility, or other areas as applicable, without the payload being moved, without evacuating personnel, and while other operations are being performed concurrently.

e. Facilities should be selected or designed to accommodate the payload prelaunch operations within a single facility or complex of connecting facilities. The movement of the payload out-of-doors between facilities is to be particularly avoided wherever possible. The storage area for payloads should be in the same facility where the prelaunch processing is accomplished to facilitate subsequent rapid minimum movement operations. Also located here should be the Tug and Orbiter simulators to avoid payload movement in the event that payload problem areas are discovered. It would be highly desirable for the payload prelaunch processing facility to be contained within the same building as the Orbiter maintenance area, or in a connected adjoining building, to provide passage of the payloads without special transportation considerations.

f. The launch base standardized test, servicing, and other support equipment should be designed and installed to facilitate operations with the several payloads being simultaneously processed in the prelaunch operation facility. Where installations are not accomplished, the equipment should be designed to be highly mobile, with the capability of coming to and supporting the payloads in their respective locations.

g. During all transportation and handling operations, the environment to which the payload is subjected should be monitored and recorded. This includes shock and vibration, temperature, relative humidity, and any other factors to which the payload is sensitive. The instrumentation and sensing equipment should be standardized in order that the data may be rapidly reviewed by standard equipment following any payload movement. The sensing equipment for the various functions may be a part of the design and installed appropriately on the payload where it may be used for other than ground transportation and handling purposes.

6. ORDNANCE CHECKOUT AND INSTALLATION

The total time required for ordnance system circuit checkout, installation of ordnance, and the installation and removal of safe and arming equipment is shown to require as much as two work shifts. Most of tne payloads incorporate ordnance and presently have the requirement for manned access into the Orbiter payload bay during operations at the launch pad. The undesirable requirement for post-mate access to the payload may be eliminated and the prelaunch operations time reduced to a maximum of one shift by incorporating the following guidelines:

a. Currently the most common method of safing and arming pyrotechnic devices involves the use of in-flight jumpers (IFJs), one of which grounds the firing networks and precludes unplanned activation of the ordnance. This is replaced before launch with a flight-configured jumper which will allow activation. These devices should be replaced by switches designed to be manually operated and with redundant interlock features. The safing/arming action may be manually accomplished on the ground or by means of a manipulator arm while in orbit. An alternate means could se remotely controlled electromechanical devices that would also possess the same fail-safe features. The design should also incorporate the capability for continuous status indication and appropriate flight crew caution and warning displays. These features will also be of use during on-orbit payload retrieval operations. Several methods have been proposed (Ref. J-11) that appear to have merit and represent the dessred approach.

b. Payloads should be designed for the installation of class A and B pyrotechnics prior to the receipt of the payload at the prelaunch processing facility. The design should include the provisions for appropriate access to the units following their installation for the purposes of possible removal/replacement of squibs and the installation of initiators prior to mating with the Orbiter.

7. TUG AND ORBITER MATING, VERIFICATIONS, AND FLIGHT PREPARATIONS

Tug mating and interface checkout, preparations for mating with the Orbiter and interface verifications, non-hazardous servicing, closeout

flight preparations, and simulated flight tests are shown to presently total as much as 21 work shifts, with 12.5 shifts as a median. The time required for each of these operations is highly dependent on the particular payload, and it is difficult to differentiate and compare the individual operations to the study goal; however, it is significant that the totals are roughly comparable to the study goal total. The goal is to accomplish these operations for any payload in a total of no more than six days, with two days allocated to the Tug operations, three days to the Orbiter simulator, and one day for mating with the Orbiter.

While the time shown as presently required is comparable to the goal, it is to be emphasized that the time for each goal operation is the maximum time allocated to be used as planned by a payload program. The operations are not expected to normally utilize all of the allocated time and the planning will include slack time for trouble shooting, module replacement, and resolution of any problem areas that might be discovered during these operations. The planning of the utilization of this time is the prerogative of the payload program with respect to the number of shifts worked, the scope and depth of testing accomplished, the operations to be performed, and the slack time felt to be required.

By utilizing the automated checkout/test equipment to conduct interface verifications and to repeat the payload status verification tests, and by conducting other operations concurrently, is expected that the Tug and Orbiter simulator operations will not require more than one day each. The subsequent mating of the payload to the Orbiter and verifications must be accomplished in approximately 12 hours, in accordance with Shuttle baseline requirements, and the additional time in the allocation for this operation is sufficient to accomplish the final closeout preparations, including the final cleaning of the payload to meet contamination control requirements.

Many of the design guidelines previously stated will also be applicable to this phase of the payload operations. The payload design should include provisions for achieving the degree of cleanliness required

commensurate with mission requirements. The requirements should be met by one or more of the following, and the approach should be in the order listed:

a. Design to tolerate the worst degree of contamination to be encountered from the factory through mission accomplishment. For example, exposed mechanisms should incorporate large clearances, loose tolerances, wipers, and seals. Solar power and thermal control surfaces should be sized to account for potential degradation from any contamination source.

b. Design provisions for self-protection or control of contamination. Examples include covers and/or gas purging, for optical or other critical surfaces, that may be actuated by remote or time command signals. The use of deflectors, baffles, traps, and sacrificial surfaces should be implemented. The payload system may include plastic bags as appropriate for the entire payload or for portions of it.

c. Design the payload to permit cleaning and to enhance the ease of achieving and maintaining the degree of cleanliness required. Examples include the selection of materials that do not generate and are resistant to the accumulation of contaminates. The materials should be compatible with cleaning solvents and the surfaces should be designed for ease of cleaning and maintaining cleanliness. The design should eliminate natural traps, inside corners, orifices, obscured surfaces, and areas that would impede the flow or puddle solvents during liquid flushing operations.

d. Contamination control, monitoring, and test equipment will be furnished at the launch base for payload use. The basic equipment will consist of a large clean booth for each payload bay in the processing facility; a centralized air source controlled to payload requirements, vacuum, flushing, and other cleaning equipment; clean-room-type clothing; monitoring; and testing instrumentation and services. Similar equipment and services will be provided for the payloads at all other areas as required.

D. CONCLUSIONS

The assumptions for the study, which consist of some of the basic low cost payload design concepts, are more than simply guidelines. They are requirements, if the goal of 10-12 days for payload prelaunch operations is to be achieved. Similarly, the guidelines developed from the examination

of the individual operations for time reduction are required, since most of them are outgrowths of the above mentioned assumptions/guidelines and show in more detail some of the ways in which the concepts may be implemented.

By implementing the 6 assumptions/guidelines and the payload and operations guidelines developed in the study, the goal of 10-12 days is possible. While the summary of the time for the payload prelaunch operations (Table J-2) indicates positive margins between the new nominal time requirements and the time allocated as available in the goal, this level is the lower limit of that which can be achieved. The margins indicated will serve to accommodate increases in time occasioned by the number of payloads in process at one time, the payload peculiars, and other variables to be encountered.

The predicted launch rates for the initial part of the Shuttle program do not require that the payload processing meet the 10-12 day goal. Also, the goal is not expected to be achieved at the outset of the program, but the capability will be designed into the system and it may be approached through an evolutionary process. The reduction in time will improve as the basic concepts are implemented and proved, the technical problems are resolved, and confidence is gained in the total approach.

Some payloads will probably utilize the 15-ft maximum diameter capability of the Orbiter payload bay and, due to size limitations during the factory-to-launch-site transportation, they will require some amount of assembly at the launch site. Since the prelaunch operations flow time is based on the receipt of a "flight ready" payload, it would appear that at least two days should be added to the flow to accommodate those payloads having this requirement. However, for a given launch base plan, this will either reduce the annual launch rate capability or require additional facilities and equipment to process at least one more payload concurrently with the others being processed. The alternative is to provide a separate on-base facility for the assembly operations and deliver the payloads in a 'flight ready' condition to the payload processing facility. The latter facility is

Table J-2. Summary of Payload Operations Time

Operation	Work Shifts			
	Existing Range	Existing Median	New Nominal	Goal Available
Receiving and Inspection (1st Day)	1-3 } 2-5	3 } 4.5	2	3
Installations	1-2	1.5		
Weight and Balance	1-2	2	0	0
Leak Testing/Propellant Loading	1-3	3	1.5	3
P/L Functional Testing	1-14	7	7	9
Transportation and Handling (Each Instance)	1	1	Eliminate some moves	-
Ordnance C/O and Installation	1/2-2	1	1	3
Preps for Mate to Orbiter and Tug	2-4	3		
Tug Mate	1-6	3		
Orbiter Mate	1-5	1.5		
Interface Verifications	1-4 } 9.5-22.5	2 } 14	12	18
Non-Hazardous Servicing	1	1		
Contamination Control	1-2	1		
Closeout Flight Preps	1-2	1.5		
Simulated Flight	1-2	1		

a highly specialized facility designed to provide a rapid uninterrupted flow of payloads to meet the launch rate requirements (10-15 day centers). The facility for payload assembly could be integrated with the facilities provided for payload refurbishment and preparations for payload return transportation to the factory or lead center. The alleviation of potential problems and scheduling effects, the shared use of equipment, and the ability to integrate the Sortie, or other large payload components coming to the launch base from different sources, are among the advantages to be gained by this method. It has been used in this study (as illustrated in Figure J-1) and is recommended for the Shuttle payload system.

To meet the goal of 10-12 days for prelaunch processing operations, a minimum of two 8-hr work shifts per day will be required. This is based on the new nominal times, which evolved from a large number of payloads, and variances in actual requirements will be encountered. It is again emphasized that the goal times allocated for each operation exceed the requirements and may be used at the discretion of the payload program.

All of the operations examined, and the associated guidelines developed, are applicable to the expendable, Sortie, on-orbit maintainable, and the ground-refurbishable payloads with two possible exceptions involving the Sortie payload. The leak testing and propellant loading operation does not appear to be required at this time. However, it is expected that the presence of pressurized Sortie modules and on-board pressure vessels will generate a requirement for similar operations and the general operations and design guidelines presented may be applied. In addition, there does not presently appear to be a requirement for ordnance operations.

E. RECOMMENDATIONS

The investigations for this study have revealed several areas which will have an impact on the time required to conduct the payload prelaunch operations, in addition to impacts on other elements of the system. Detailed studies are recommended for the following:

1. The operations involved in handling radioactive thermoelectric generators (RTG) for payloads using this equipment as a power

source present problems for which there have been no really satisfactory solutions developed. The problems of thermally cooling the units on the ground and protecting personnel from radioactive exposure in unmanned systems is difficult. Their use in the Shuttle may be expected to be more difficult, particularly where the system will be required to be installed late in the flow and most probably on the launch stand.

2. It is always highly desirable that a payload arrive at the launch base as a completely integrated "flight ready" unit and, since it is inevitable that some payloads will approach the maximum limits of 4.6-m diameter by 18.3-m length, a considerable problem exists in how to transport the payload from the factory to the launch site. Diameter is particularly important, since it is usually the controlling factor in any of the modes of transportation. Today the few outsized units are handled by special road permit, in Guppy-type aircraft, or by a barge waterway system, all of which are expensive. Presently, the C-5A cargo aircraft can handle 4.1 m high x 5.8 m wide x 37.0 m long cargo and can exceed Orbiter payload weight limits. While the Guppy can handle the maximum payload diameter, there are critical weight/length limits and few aircraft are available. The requirements for the transportation of large size payloads will increase in the Shuttle era, and the solution to the problem could result in increased assembly/test operations and facilities at the launch site. Since this problem area does not appear to have been addressed by any study, it is recommended that a detailed study of payload transportation be initiated. Consideration should be given to all movement requirements, including the transportation from an alternate landing site to the required destination. A trade-off study should be conducted to determine the effects of one-piece vs many-piece payload designs with respect to cost effectiveness, considering all elements of the payload systems including assembly, test, and on-orbit and refurbishment operations.

3. The payload ground-refurbishment cycle needs to be studied in detail to resolve many problems/questions that have appeared. For example, should the refurbishment operations be conducted at the launch site, the depot, or the factory? If conducted at the launch site, will the test philosophy for the "new" payloads be different from the original, with respect to thermal vacuum, acoustic, or other acceptance testing, or will facilities for this testing be required at the launch base?

4. The solution to the potential problems of compatibility of software developed for use at the factory and that to be used

at the launch base for testing purposes should be resolved. The use of universal test equipment stations located at the factory versus other methods should be examined for cost effectiveness. The impact of implementing the reduced testing concept at the launch base should be evaluated to determine the extent to which software costs may be reduced.

5. The on-orbit payload repair/maintenance should be studied in detail for each of the applicable payloads. The study should determine requirements, capabilities, and limitations and examine the impacts on ground testing and other objectives where compatible multiple purpose features may be incorporated in the payload design.

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APPENDIX K. LAUNCH SITE TESTING

A. INTRODUCTION

Present-day launch site testing is a repetition of the factory integrated system test (IST) on most satellite programs. For some programs a complete set of factory AGE is shipped to the launch site, where it is set up to run the IST. On other programs, a complete set of AGE is installed permanently at the launch site where it is again used to run the IST. After the satellite is delivered to the launch site and inspected visually for in-transit damage, an IST (perhaps abbreviated) is run to assure no electrical failures. The satellite is then mated to either the transtage or booster adapter fixture and made ready to mate to the booster. Usually an IST is run at this point to assure proper performance prior to mating with the booster. After being mated with the booster, placed on the launch pad, another IST is run as near as possible to the start of the countdown. This ends the test sequence at the launch site with the exception of anomalous conditions. It must be pointed out that the above sequence represents generally the procedure presently followed for the medium to large classes of satellites. There are some variances to the procedure described, but generally it is representative of most satellite programs.

Upon inspection of the above sequence, it becomes obvious that the time the average satellite is actually in test (typically 30-50 hours) represents a small cost impact when compared to the cost of the AGE hardware, the expense of transporting the test and maintenance crews to the launch site, and the expense of retaining the test and maintenance crews at the launch site for extended periods. For a typical launch operation, approximately 20 to 40 man months of effort are expended on operation and maintenance of the test equipment. If testing at the launch site requires an extra set of electrical AGE, test equipment, computer (not including software), communication and power equipment, and attitude control, it can add from 1.5 to 3.0 million dollars to the project cost.

It can be seen that any effort to save cost by reducing the amount of testing would not reduce cost. Effort to develop simpler test equipment to be used exclusively at the launch site, would cost more than it would save. Hence, saving must come from the elimination of launch site testing, rather than reducing it. This approach assumes the Shuttle has the capability to perform relatively low-level payload tests. The probability of success is great enough to offset the risk involved in not performing a full integrated system test. Further, it must be shown how much a failure would cost versus the savings realized from the elimination of launch site testing.

B. ASSUMPTIONS

The capability to test the payload before deployment involves supplying power to the payload, communication with the payload, and data processing and display of test results. Each payload contractor will be required to supply software necessary to conduct the test from the Shuttle computer.

C. POWER REQUIREMENTS

If prelaunch and pre-orbit tests are to be conducted by the Shuttle on its payload, that payload must first be supplied by the Shuttle with sufficient power properly conditioned to suit each payload. This requirement could be made easier if a standard bus voltage could be agreed upon for all satellite contracts, since it is assumed this standard would also be adopted for the Shuttle power system. In any case, power, properly conditioned if necessary, will be assumed to be available from the Shuttle.

D. COMMUNICATIONS REQUIREMENTS

It will be assumed here that docking and undocking operations will be controlled from the Shuttle. This requires a communications link between the payload and the Shuttle for the purpose of commanding the attitude of those satellites to which the Shuttle will dock on orbit. A further

assumption is that a checkout of the payload will be required prior to injecting it into orbit. This again requires a link between the payload and the Shuttle, along with a computer to process the telemetry data. There are several methods for presenting the telemetry data to other systems, but for purposes of this document it will be assumed the Shuttle will contain a telemetry decommutator, a frame synchronizer, and a word formatter. From this equipment, data will be presented in a synchronized, parallel format with enough control lines to indicate when a new data word is present, when telemetry word one is present, and when the telemetry is synchronized (locked). The maximum data rate considered will be 256 kilobits, based on Reference K-1.

E. DATA PROCESSING REQUIREMENTS

In line with the preceding assumptions made for the communications link, there will be a computer available on the Shuttle. This computer will be used to process downlink telemetry and uplink commands. For those payloads whose command link is interactive, the computer will provide the closed loop necessary to execute commands.

This computer is assumed to be for general usage in the Shuttle and one of its tasks will be processing telemetry for checking out the payload both on the launch pad and immediately prior to orbit injection. In order that the computer handle the highest specified telemetry rate (256 kilobits) it must exhibit at least the following characteristics:

16 bit word length (minimum)

65,536 words of executable memory

10 million words of mass memory

Execution rate of 400-500 kilo-instructions per second (KIPS)

A computer meeting these minimum general requirements would be classified a mini-computer and would be capable of processing telemetry at the maximum rate, but very little interactive testing could be accomplished.

The general type of testing possible with a system of this nature is limited to activating a subsystem and checking its status. With the limited data processing capability described above, other systems could not be checked for spurious reactions and data trends could not be checked. These are generally classified "go, no go" low-level tests.

F. DISPLAYS REQUIREMENTS

For the low-level type testing assumed here, the display requirements will be relatively slight. A Cathode Ray Tube (CRT) display with graphic capability, interactive keyboard, and light pen will be required. The CRT should be capable of displaying 40 to 50 lines of 80 characters each. This display must be available to the payload specialist who will act as test conductor. For the most part, the tests anticipated will be performed in prearranged groups, each of which will be started from the keyboard and the results (sometimes graphic) will be displayed on the CRT. An integral paper tape reader could be used advantageously to automatically run tests, but is not considered a requirement.

G. ANALYSIS

1. PRE-ORBITAL TEST REQUIREMENTS

Although testing prior to orbit injection has been assumed, the level and intent of that testing has not been specified. The level of testing will be constrained by the capability of the data processing equipment, so the question becomes how much testing is required so that adequate data processing can be provided.

The tests performed just prior to orbit injection need not be extensive for the expendable and refurbishable payloads because the detection of a failure is all that is required and not the isolation or diagnosis of that failure. Since these types of payload are not serviceable in space, nothing could be done for a failure even if it could be diagnosed. The important fact is that there has been a failure in the payload and it can be returned for ground repair.

For the on-orbit maintainable payloads, it is unlikely that spare modules would be carried into orbit with the payloads because of volume limitations. Diagnostic testing, then, is not a requirement, since the information is of no use to the Shuttle mission even if it is available. Low-level status testing is all that is required so that each subsystem is checked to assure it has not failed. The mini-computer described in general terms in Section E is completely adequate for this type of testing. This also simplifies the software required in the Shuttle computer, since complex cross checking and data trending is not required.

2. BASELINE CONFIGURATION

A baseline test configuration and four alternates to the baseline have been developed for comparison purposes. The baseline represents generally the present method of launch site testing for medium and large satellites. Each of the alternates will be considered for its impact, both technical and economic, to the baseline for the four missions. Note that for each of the five configurations it is assumed that the payload is delivered to the launch site fully tested and qualified for launch. No recommendations on factory testing are being put forth here and only the subject of launch site testing will be considered. Table K-1 is a matrix comparing each of the alternate configurations.

The baseline, as stated previously, represents generally the present methods of launch site testing and preparation for medium and large satellites. A complete set of factory AGE is supplied at the launch site for testing the satellite. Integrated System Tests (IST) are performed on the satellite in the Satellite Assembly Building prior to mating with the booster and again (after mating to the booster) just prior to the start of the countdown on the launch pad. No checkout capability is provided on the Shuttle. The satellite is launched, placed in orbit, and checked out from its ground station. In case of an anomaly, a decision would be made at this point on whether to re-dock the satellite to the Shuttle and return to earth, leave the satellite in orbit to be repaired on a subsequent mission, or discard the satellite as a failure.

Table K-1. Alternate Test Configurations

	FACTORY AGE	GROUND AGE AT LAUNCH SITE	SMALL COMPUTER IN SHUTTLE	LARGE COMPUTER IN SHUTTLE	COST SAVED OVER BASELINE	RISK
BASELINE	X	X				NOMINAL
I	X	X	X		NONE	LOW
II	X		X		10 %	HIGH
III	X			X	SPECIAL COMPUTER DEVELOPMENT	
IV	X		X		NONE	

K-6

3. ALTERNATE CONFIGURATIONS

a. Alternate I

This alternate represents the baseline with the addition of low-level checkout capability aboard the Shuttle. This requires the installation of a mini-computer aboard the Shuttle such as described in Section E. The sequence of operations is the same as for the baseline up to and including the IST run just prior to the start of the countdown. At this point, the Shuttle would run a short (1-4 hours) status test on the satellite. After launch and prior to orbit injection, the Shuttle would repeat this test. If an anomaly is discovered at this point, the satellite could easily be returned to earth since it has not yet been undocked from the Shuttle.

b. Alternate II

This alternate is an extension of the baseline and Alternate I; Alternate II is Alternate I without the factory AGE.* Alternate II requires the mini-computer but does not require factory AGE.* The IST runs prior to mating with the booster and prior to countdown under Alternate I are eliminated. The only prelaunch test is the status test run prior to countdown from the Shuttle. After launch, the operation is the same as described in Alternate I.

c. Alternate III

Alternate III is a major departure from the baseline in that it requires no factory AGE* at the launch site but requires a large computer in the Shuttle which can perform the same tests on the satellite as those performed by the factory AGE. The satellite is delivered to the launch site fully tested and qualified for launch. It is then mated to the booster and a full IST is run using the Shuttle computer. After launch and before orbit injection, another IST (probably abbreviated) is run from the Shuttle. If an anomaly is discovered, the satellite can then be returned to earth.

* Electrical AGE

This alternate requires a rather large computer aboard the Shuttle in order to perform ISTs on the large satellites. This computer must be generally of the class of an IBM 370/145 (proposed by IBM for the DSP Phase II AGE) with approximately 512 K bytes (128 K words) of executable memory and 100 million bytes of mass memory.

d. Alternate IV

This alternate represents the largest departure from present methods, in that it requires all satellites to include complete onboard checkout of each subsystem on the satellite. This would include small computers (Micro class) built into each subsystem which constantly run diagnostics on that subsystem and report status to a central computer for compilation.

The satellite, as in all the other cases, would be delivered to the launch site fully checked out and qualified for launch. No testing would be performed as such, but after mating to the Shuttle the satellite would be connected hardline to the Shuttle computer, which would monitor the performance of the satellite subsystems through the onboard checkout units. This is equivalent to constant testing while the satellite is turned on; hence, it would satisfy the pre-orbital checkout requirements also.

4. OPERATIONAL AND TECHNICAL IMPACTS

The technical impact to present-day operations will vary with the alternate configuration. The baseline causes zero impact to present-day methods, since it represents the general methods now followed.

Alternate I presents very little impact to present-day operations, since the only requirement above the baseline is for a small computer on the Shuttle which will have access to the telemetry decommutators. This computer will be available and only the interface with the decommutators must be added (assuming the telemetry receivers and decommutators are available). Additional software would be required from the contractor to run the tests in the Shuttle computer. This software would be a functional subset of the software developed for the factory AGE computer, but since the likelihood of having

program-compatible computers is very slight, the software would be re-written. This software would be a requirement for any pre-orbital tests performed by the Shuttle also, so it cannot be completely attributed to pre-launch test requirements.

Alternate II seemingly presents a large impact to prelaunch test operations, but this impact is probably more emotional than technical. The question to be answered here is "How essential is a complete prelaunch test?" This alternate configuration eliminates the IST-type test and substitutes a short status-type test just prior to launch. As in Alternate I, this presumes the availability of a computer which has a telemetry interface, so there really is no technical impact over that previously mentioned of needing the interface between the computer and the telemetry system. Operationally there is major impact, in that no ground testing as such is conducted using ground AGE.

Alternate III would have major impact, both operationally and technically. Since this alternate calls for a large computer aboard the Shuttle to run an IST-type test before launch, it also requires test personnel on the Shuttle to conduct the tests. Technology has not advanced as yet to the point that such a large computer qualified for space is available, so it would also require a major advance in the state of the art of spaceborne computers.

Alternate IV would have little impact to Shuttle operations or technology, but represents a major impact to the design and manufacture of satellites. It would require the contracting agency to specify a type system which has not as yet been developed. Each contractor would be given the latitude of developing his own onboard checkout techniques and equipment but each would be required to meet a standard interface. The state of microcomputers has not as yet been advanced to the degree necessary for this application, but there are several very small, space-qualified computers from which the technology could be developed.

5. ECONOMIC IMPACTS

In order to bound any cost savings made possible by reduced scope of launch site testing, some costs must be attached to current test operations. These costs would be associated with the baseline configuration. The costs are developed for three classes of satellites:

a. Large satellites launched singly and generally of the refurbishable or maintainable type.

b. Medium size satellites launched in groups of two or more and likely to be of the expendable type.

c. Small payloads often of the "piggy back" class.

The cost of test operations at the launch site is attributed to two major contributors.

First, there is the cost of transporting the test crew and AGE maintenance personnel to the launch site and back to the factory and supporting them while they are at the launch site. This can involve from 5 to 15 men for up to two months.

The second major contributor to launch site costs is the cost of supplying a duplicate copy of the factory AGE* for use at the launch site. The cost for factory AGE varies with the size, complexity, and type of satellite, but the major determinant of cost of the AGE* is the data processing equipment. Generally, AGE* costs range from $1 million to $3 million with a small number of programs of special nature being outside these limits on both sides (software cost not included).

From this, the cost of launch site testing can be derived by adding a fixed cost (operations) to the cost of a set of factory AGE* amortized over the number of launches scheduled. For purposes of this document, the number of launches will be assumed to be 5. For a specific mission, the launch site testing cost could be computed exactly, but average costs will be used here. An average of $5,000 per man month will be used for operations costs.

* Electrical AGE

a. Baseline Costs

Large Satellites

Operations, 30 man months	150,000
AGE Hardware*	600,000
Total Cost Per Launch	$750,000

Medium Satellites

Operations, 20 man months	100,000
AGE Hardware *	400,000
Total Cost Per Launch	$500,000

Small Satellites

Operations, 15 man months	75,000
AGE Hardware *	200,000
Total Cost Per Launch	$275,000

b. Alternate I Cost

Since Alternate I does not eliminate any of the baseline testing, there will be no cost reduction. Instead, there will be a slight increase in the cost due to the added requirement of programming the mini-computer aboard the Shuttle. This cost is not considered significant, however, and for purposes of comparison Alternate I cost will be the same as baseline.

c. Alternate II Cost

Since Alternate II proposes eliminating launch site testing, using factory AGE*, all the baseline costs would be saved. No added costs can be attributed to this alternate with the exception of the previously mentioned software cost for the Shuttle computer. The Shuttle software cost, again, is not being considered since it is relatively small and is associated with another requirement (pre-orbital tests).

* Electrical AGE

The percentage of launch cost which can be saved using this alternate is:

Large Satellites	7.1%
Medium Satellites	9.5%
Small Satellites	13.1%

This assumes a Shuttle launch cost of $10.5 million and the following additional assumptions:

(1) Single large satellites are launched with one medium satellite.

(2) Medium satellites are launched in groups of two.

(3) Small satellites are either "piggy back" with other missions or launched in effective groups of five.

d. Alternate III Cost

Alternate III requires no launch site testing using the factory AGE*, but it requires an equivalent to the IST to be conducted from the Shuttle using the Shuttle computer. The costs associated with duplicating the AGE will be saved, but software to run the test in the Shuttle computer will be required. This software is functionally identical to that already developed for the factory AGE*, but it must be rewritten for use in a different computer (the Shuttle computer). It is assumed this software will cost half as much as a set of factory AGE, so half the hardware cost will be saved. Since no AGE* maintenance personnel are required, and it can be assumed that the Shuttle equipment will be more compact and easier to operate, the operations cost will be approximately half those for the baseline. Therefore, this alternate saves approximately half the baseline cost but requires a very large (and expensive) computer on the Shuttle.

e. Alternate IV Cost

Since this alternate conducts no ground test using the factory AGE*, all the costs of the baseline are saved. There would, however, be added cost

* Electrical AGE

on the satellite of designing and installing the onboard checkout devices. Probably the net effect on the smaller satellites would be to increase project costs, but it would be impossible to quantify these costs at this time.

H. RESULTS AND INTERPRETATION

Of the configurations, the baseline and Alternate I offer the lowest risk and highest confidence of successful operation. The baseline is representative of current operations and has been developed from many years of experience. Alternate I includes all of the baseline and adds the further dimension of conducting a status-type test from the Shuttle. The advantage of these configurations is that they require little new procedural or technological development. The main disadvantage of these alternates is they are relatively expensive, although the cost of launch site testing does not represent a significant portion of the total launch cost.

Alternate II offers the greatest cost savings but presents some element of risk in that no detail testing is performed after the satellite leaves the factory until it is on orbit. This requires a high confidence in the satellite in order to launch without a detailed test at the launch site. A degree of confidence can, of course, be gained by running the low-level status tests from the Shuttle prior to launch.

Alternate III reduces the risk, but requires a large computer aboard the Shuttle. This allows the detailed tests to be run before launch, but they are conducted from the Shuttle without the need of factory AGE. The disadvantages are that a set of extensive software must be developed for the Shuttle computer and that such a computer is not available using present technology. The size of the computer is the real problem because there is no demand for data processing equipment of this nature. Thus, it would require special development which would make it prohibitively expensive.

Alternate IV, while offering great advantages in all aspects of testing, seems somewhat utopian at this time. If each of the subsystems of a satellite could be made to completely test itself, this capability could be used

even down to the assembly line level. This approach would require a great deal of research and, even if feasible, it would have to be required for all satellites so that standardization could be effected. Until these subsystems become widespread, they would also be very expensive.

I. CONCLUSIONS

It appears, without extensive study of the subject, that Alternates III and IV are inadvisable at this time. If the state of the art of either large spaceborne data processing systems or self-contained diagnostic equipment were advanced significantly in the near future, these could become attractive. For the present, they are unattainable.

The baseline does not include a capability which is required; namely, the ability to perform a satellite checkout prior to orbit injection. There are methods of circumventing this restriction (e.g., relaying the satellite telemetry to the ground for checkout), but the requirement is not difficult to meet using a computer already installed on the Shuttle. There is also the need for command and control contact between Shuttle and satellite whose requirements overlap those of testing.

The choice, at this time, appears to be between Alternates I and II, making it a choice between testing for higher confidence or accepting a greater risk for cost saving. The smaller satellites appear to offer a good trade-off between these factors requiring relatively low confidence factors, but the large satellites whose failure would abort the complete mission may not be a good vehicle for this cost savings.

J. RECOMMENDATIONS

Alternate II is recommended as the optimum trade-off between cost savings and risk for satellites which are not launched singly. A further study of the incidence of launch site failures and the level of testing possible with Shuttle equipment is suggested tc make a final selection of the mode of operation. Further study should also pursue the questions of spaceborne data processing equipment and the viability of requiring self-contained diagnostic

equipment. The availability of either Alternate III or Alternate IV could change the recommendations.

Further study should be pursued on aspects of testing other than at the launch site. The modularity of the repairable and refurbishable satellites could yield some cost savings if the test equipment could also be standardized. Test sets such as those developed for integrated circuit testing could present opportunities for reducing both assembly line testing and integration testing. These test sets would include a computer and be programmable over a wide range of subsystems and could be integrated into a larger test set for system testing.

The use of Universal Test Equipment (UTE) at both the factory and the launch site should also be studied to determine its feasibility. The greatest problem in using UTE is designing it to accept the largest, most complex satellite and yet be cost effective on the smaller satellites. Wide usage of UTE would be required in order to make it cost effective, and if the software required for its use is more costly than developing special AGE, then the UTE may not be economically feasible.

K. REFERENCES

K-1. Space Shuttle Baseline Accommodations for Payloads, NASA MSC 06900, Johnson Space Center, Houston, Tex. (27 June 1972).

APPENDIX L. DESIGN

Analyses have been conducted to identify alternative approaches to the modularization of payloads, to select preferred approaches where possible, and to identify the considerations or tradeoffs necessary to arrive at practical solutions to the many problems that will be encountered. The analyses have been conducted by establishing a conceptual design for each of two diverse types of payloads which are used as baseline or reference designs. The Defense Support Program (DSP) spacecraft was used to represent a typical synchronous orbit payload launched and serviced on orbit by the Tug. Most of the conceptual design effort on DSP is presented in Reference L-1. For completeness, the work is included herein when appropriate. The Large Space Telescope (LST) was used as a baseline vehicle representative of relatively large, low-earth-orbit (LEO), Orbiter-launched and serviced payloads. A conceptual design of a modularized Earth Observatory Satellite (EOS) developed under NASA Study 2.6 was also considered for reference purposes.

The analysis of the subject Serviceable Payload Modularization is presented in sections under the following subject headings:

A. Replaceable Module/Payload/Service Module Compatibility
B. Replaceable Module Thermal Control
C. Replaceable Module/Payload Fluid Interfaces
D. Replaceable Module/Payload Electrical Interface
E. Non-replaceable Equipment in an On-Orbit Maintainable Payload
F. Module Removal/Replacement Using Manipulator
G. Module Backup - Handling Provisions
H. Recommended Tradeoff Studies

It was assumed that the design problems encountered with the DSP, LST, and EOS would be typical of those encountered in the design of any payload for on-orbit maintenance.

A. REPLACEABLE MODULE/PAYLOAD/SERVICE MODULE COMPATIBILITY

The design of a payload for on-orbit servicing should not be divorced from consideration of the type of motion selected to transfer replaceable modules to and from the payload, the design of the service module, and the design of the modules themselves.

The transfer motion selected should be one which will result in simple, reliable, lightweight mechanisms to accomplish the transfer and at the same time result in a spacecraft configuration which is compatible with the requirements of the mission. The service module design should, of course, be compatible with the transfer motion selected and the spacecraft configuration.

To help illustrate the design process for a space serviceable spacecraft and its servicing equipment, a discussion of the design of the Space Serviceable Defense Support Program (SSDSP)* spacecraft is presented in Reference L-1.

The SSDSP was assumed to be typical of spacecraft in geosynchronous orbit. Furthermore, it was assumed that the general nature of design problems encountered with the SSDSP design would be typical of those encountered in the design of any payload (low-earth to geosynchronous orbit) for on-orbit servicing. Hard-docking between the Tug and the spacecraft was assumed in all cases to accomplish on-orbit servicing.

The study presented in Reference L-1 concluded that the design of a spacecraft for on-orbit servicing should not be divorced from the design of the service module, the design of the modules, and consideration of the type of motions required to achieve transfer of the modules to and from the spacecraft.

The transfer motions selected should result in the design of a lightweight, simple mechanism to achieve those motions.

The shape of the spacecraft and the service module should be compatible with the transfer motions.

* This program is also referred to as the Shuttle-Launched Defense Support Program (SLDSP).

The module design should allow for simplicity of removal/insertion, alignment and locking in place. Furthermore, it should be a basic design common to all modules in the spacecraft and be able to meet the packaging requirements of each of the modularized subsystems.

B. REPLACEABLE MODULE THERMAL CONTROL

Design studies of the Defense Support Program (DSP), Earth Observatory Satellite (EOS), and the Large Space Telescope (LST) spacecraft were configured for on-orbit servicing via a remote servicing module have indicated that the thermal control system design of the individual modules should be self-contained.

Ground refurbishable or expendable spacecraft which have no replaceable modules that must be automatically positioned and therefore no movable module/spacecraft interface do not require self-contained thermal control systems for each subsystem or grouping of subsystems.

The three payloads (DSP, EOS, and LST) chosen for the space serviceable design studies were assumed to be typical of spacecraft in geosynchronous and low-earth orbit. It was assumed that the design problems encountered with these spacecraft would be typical of those encountered in the design of any spacecraft for on-orbit servicing.

The design studies performed and reported on in Section 4 of Volume II and Section 3 of Volume III were performed for the on-orbit maintainable type of payloads. In these design studies, an interface between the module and the spacecraft which would be typical of that required for an on-orbit serviceable spacecraft was designed. This interface design is shown in Figure L-1. When the module is locked into position on the spacecraft, it contacts the spacecraft at three points only. These three points are indicated by the arrows on Figure L-1. The main feature of this interface is that it is simple and has a minimum of contact points. However, the concept of minimum contact points is not compatible

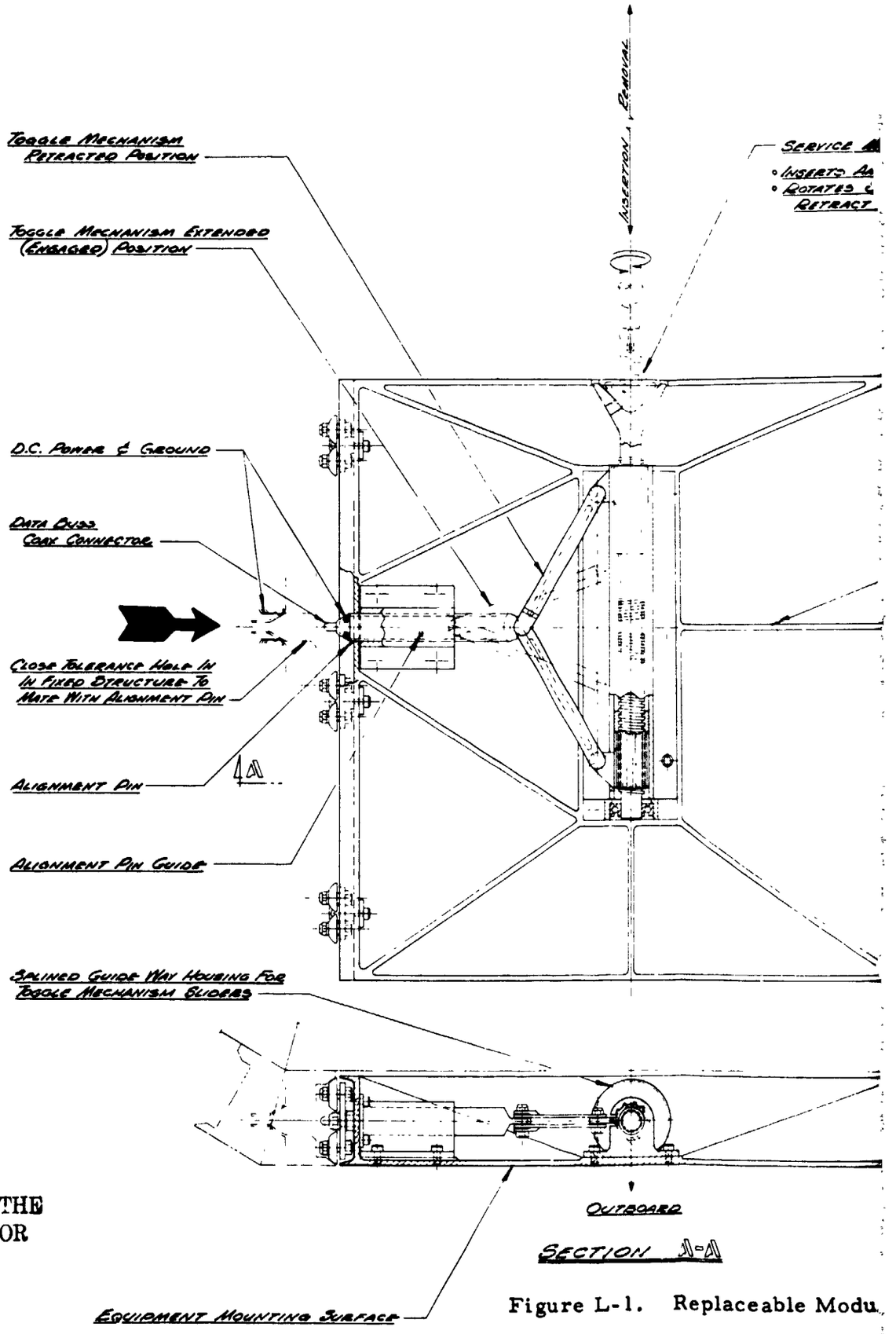
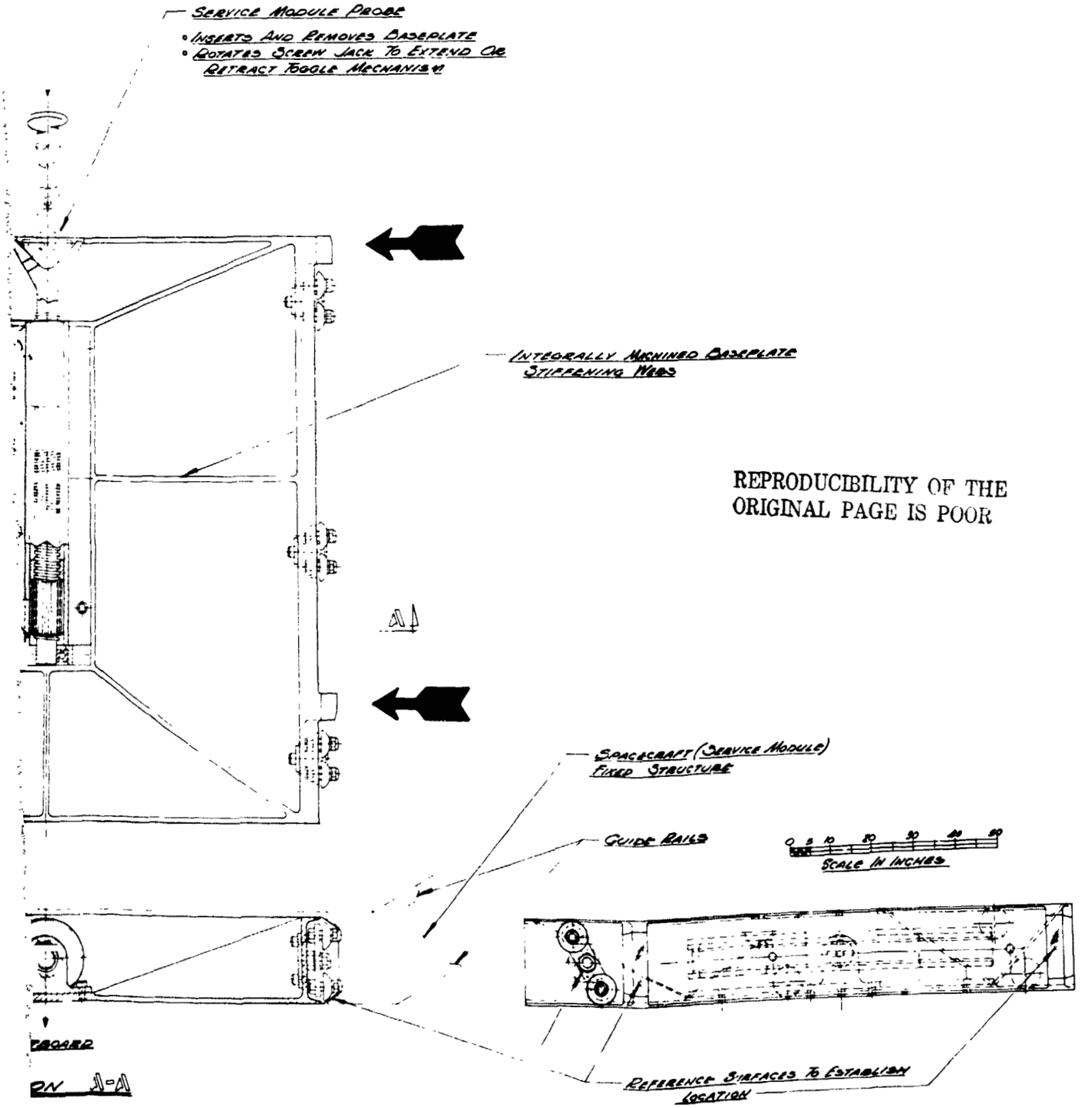


Figure L-1. Replaceable Modu

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1. Replaceable Module Baseplate

FOLDOUT FRAME

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with the transfer of heat across the interface through conduction. Minimal contact between the module and the spacecraft does not provide good conductive heat transfer, therefore, it is desirable to have a self-contained thermal control system. In addition, conductive heat transfer across joints quite often requires high clamping pressures (15-20 x 10^5 N/m^2) and very smooth surface finishes in order to minimize thermal resistance. High clamping pressures are difficult to achieve with a joint designed to be mated and demated in space. Also, the possibility of cold welding the joint could be a problem.

Because typical sensors for spacecraft suite often include those which operate in the IR region, cooling to temperatures in the neighborhood of 90°K - 120°K could be required. For example, following is a partial list of IR sensors for the EOS mission, along with their required detector operating temperatures:

1.	Thematic Mapper (7 Detectors)	290°K 290°K 290°K 290°K 120°K 120°K 90°K
2.	Cloud Physics Radiometer (InAs Detectors)	120°K

As can be seen, at least four of the detectors must operate at 120°K or lower (Reference L-2). Another example is the focal plane detector array for the DSP which also must operate at relatively low temperatures. Achievement of operating temperatures in these ranges requires either the use of second surface mirror radiators, solid cryogen refrigeration or some type of active refrigeration system. Furthermore, to transfer the heat away from the detectors to the refrigeration system or radiators, some form of conduction path, such as a conduction bar or heat pipe is required. In incorporating these concepts into a space serviceable spacecraft design, two basic approaches can be considered:

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a. The cooling system can be separated from the detectors such that a joint capable of being demated and mated at the module/spacecraft interface is required in the conduction path between the detectors and cooling system, or

b. The entire system can be self-contained within the detector module.

The conduction of heat across a joint as in choice (a) above requires very smooth surface finishes and high clamping pressures to minimize the ΔT across that joint. Experience with the second surface mirror radiator/conduction bar cooling system for the DSP has shown that, even with a joint between the conduction bar and radiator which has been assembled on the ground and is not intended to be separated, it is difficult to achieve good thermal conductivity across that joint. The DSP utilizes a magnesium conduction bar joined to a magnesium radiator frame to which second surface mirrors are mounted. The two are joined via four 10-32 screws. Heat is conducted away from the detectors, through the conduction bar, across the joint and through the radiator to space. Flight experience with the DSP had shown that the detectors were not being maintained at their operating temperatures due to a reduction in preload of the screws holding the radiators to the conduction bar. The solution to the problem was to specify very smooth surface finishes (10 microinch RMS) and a clamping pressure of 20×10^5 N/m^2 minimum. The clamping pressure was monitored during assembly via strain gages mounted on the bolts. Other methods that can be used for achieving good thermal conductivity across joints include the use of indium foil or thermal grease combined with high joint pressures. Figure L-2 shows the variation of contact thermal resistivity with joint pressure for a variety of materials and surface finishes.

The foregoing discussion shows some of the requirements for achieving a good thermal conducting joint. Attempts to achieve good conductivity with a joint which must also be mated and demated in space would result in difficult design problems. The high clamping pressures and

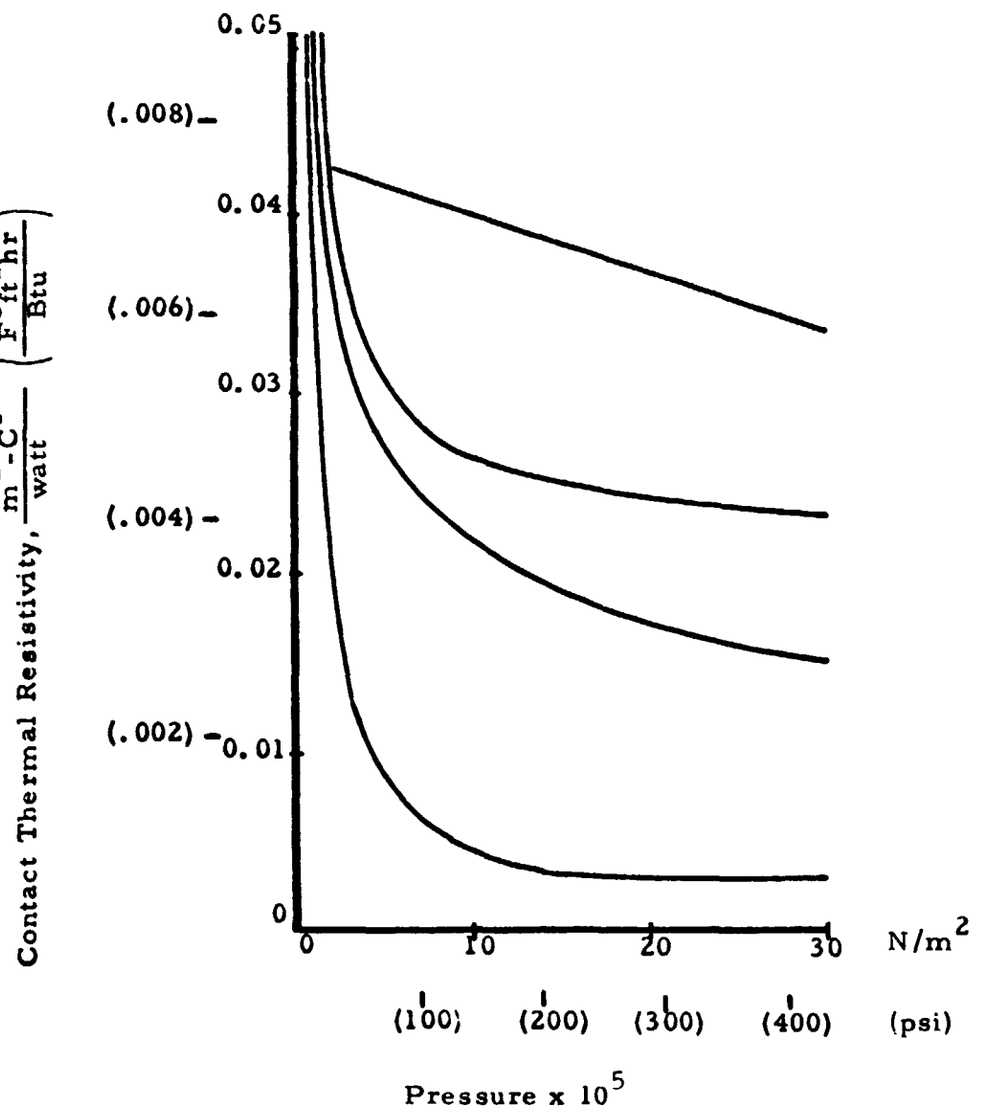


Figure L-2. Contact Thermal Resistance

very smooth surfaces required to achieve good heat transfer are, on the other hand, conducive to cold welding in a vacuum. The mechanics of a mechanism which would provide high clamping pressures, even if the cold welding problem could be solved, would be very complicated.

For the above reasons, it is felt that the thermal control system for each module should be self-contained (choice b). The design of a self-contained thermal control system for each module can be readily achieved. For the SSDSP spacecraft, the thermal control system design is reported on in Reference L-1. Each module has either a passive system such as multilayer blankets or mylar or second surface mirrors which act as passive radiators, or active control such as heaters. Heat is transferred within the module via conduction bars or plates or heat pipes. Although the SSDSP did not utilize active cooling such as solid cryogens or Vuilleumier refrigerators, this could be readily accomplished. The replaceable feature of the modular concept would make active cooling, where required, an attractive concept.

The thermal control system for the EOS is quite similar to that of the SSDSP, except that the many IR sensors on board the EOS require some form of active cooling to achieve the $90^{\circ}K$ - $120^{\circ}K$ detector temperatures. Here again, the replaceable feature of the module makes it attractive to consider active cooling.

C. REPLACEABLE MODULE/PAYLOAD FLUID INTERFACES

Fluid interfaces between a module and the payload which require demating and mating each time a module is replaced should be avoided in the design of a space serviceable spacecraft by making all fluid systems self-contained within a module.

Design studies (Section 4 of Volume II and Section 3 of Volume III) in which the Defense Support Program (DSP), Earth Observatory Satellite (EOS), and the Large Space Telescope (LST) were configured for on-orbit servicing via a remote service module have indicated that it is desirable

to maintain the interface between the module and the payload or spacecraft as simple as possible. The interface should consist only of the mechanical interface required to align the module to the spacecraft and carry flight loads, and the electrical interface required for power and data transmission. Addition of a fluid interface would complicate the module/spacecraft process by requiring at least one extra joint which would have to be aligned prior to the final operation of locking the module to the spacecraft or require a mechanism which would align and lock the mating halves of the connector after locking the module to the spacecraft.

Ground-refurbishable or expendable spacecraft which have no replaceable modules could be designed to have separable connectors, although generally, it is recommended that brazed connections be used whenever possible.

The difficulties involved with mating and demating fluid connectors remotely in space are influenced by the type of fluids required to be transferred. The configuration of the spacecraft and the type of on-board subsystems will determine the fluids used. These fluids could range from gases such as helium or nitrogen to room temperature liquids such as hydrazine to cryogenics such as LH_2. The launch environment, orbit, orbital maneuvering requirements and reentry environment determine the conditions under which these fluids must operate. A partial list of some possible subsystem operating environments is shown in Table L-1. As can be seen from this list, the operating environments could range from cryogenic temperatures to temperatures well above room temperature, depending upon the fluids used. Thus, if we consider the use of mateable/demateable connectors, the problem becomes one of designing connectors capable of transferring the above-mentioned types of fluids with negligible leakage over a wide range of temperatures and pressures. In addition, the mechanisms for achieving the connection remotely in space must be designed. Once the connection is made, a method of checking for leaks must be employed.

Table L-1. Possible Subsystem Operating Environments

Subsystem	Medium	Temperature	Pressure
Pneumatic Control	He, N_2	256 - 348° K	3450/314 x 10^4 N/m^2
Fuel Tank Pressurization	He, N_2	256 - 348° K	138 - 414 x 10^4 N/m^2
Oxidizer Tank Pressurization	He, N_2	256 - 348° K	138 - 414 x 10^4 N/m^2
Oxidizer Tank	N_2O_4	278 - 348° K	138 - 414 x 10^4 N/m^2
Fuel Tank	N_2H_4	278 - 348° K	138 - 414 x 10^4 N/m^2
Cooler Tank	He, H_2	55 - 110° K	0.17 x 10^4 N/m^2
Vacuum Monitoring	Vacuum	-	-

Reference L-3 reports on the experience gained in the development of over 1,300 separable (but not remotely mateable/demateable) connectors for the Saturn IV stage. The problems encountered over a long development program were manifold and complex. After installation of the connectors by highly trained and skilled personnel, many leaks were uncovered at the system checkout level. Table L-2, taken from Reference L-4, shows the number of leaks found at the system check level.

Table L-2.

C/O	Approx. Number of Leaks	Approx. Number of Connections
1	40	500
2	80	800
3	Not Available	900
4	110	900
5	70	1,000
6	70	1,100
7	90	1,200
8	60	1,300

It can be seen that, at times, over 10 percent of the connectors showed leakage, even after being assembled with great care in clean conditions by trained personnel. It is reasonable to assume that the potential percentage of leaks encountered in a system designed for remote coupling in space would be at least this great and probably greater.

To detect leaks, however, would require on-board leak check equipment. In all probability, more than one type of leak detection system would be required. In a discussion of methods of static-leak detection in

Reference L-5, it is pointed out that no single method is used because none is completely universal. The most suitable method for a given situation depends on a number of criteria:

1. Fluid used for pressurization
2. Magnitude of the anticipated leakage
3. Configuration of the hardware to be checked
4. Environment of the leak; i.e., temperatures and pressures involved
5. Accuracy required
6. Whether it must be quantitative or qualitative
7. Time and facilities available for this effort.

The methods of leak detection used by Rocketdyne are shown in Table L-3 which is taken from Reference L-5.

Table L-3.

	Measurement Capabilities					Practical Applications					
Method	Liquid	Gas	Quantitative	Qualitative	Practical Lower Limit scc/sec	Joint Development	Component Checkout	Engine Checkout	Component Ground Test	Engine Ground Test	Flight Test
Soap Solution		X		X	10^{-1}			X			
Pressure Lockup		X	X	X	10^{-2}			X			
Liquid Submersion		X		X	10^{-1}		X				
Rotameter		X	X		10^{-1}	X	X	X			
Mass Spectrometer		X	X		10^{-10}	X	X				
Sight Glass	X		X		10^{-2}				X	X	
Balloon		X	X		10^{-1}				X	X	
Water Displacement		X	X		10^{-3}	X		X	X	X	
Burst Diaphragm		X		X	10^{-2}				X	X	
Hastings-Raydist Flowmeter		X	X		1	X			X	X	X

It was also noted in Reference L-5, that the Hastings-Raydist Flowmeter, the only instrument listed for in-flight leak detection is "a proposed, but not a proven method".

Reported on in Reference L-6 are a number of sensors capable of detecting gas leaks. They are divided into three groups: hydrogen sensors, oxygen sensors, and non-specific sensing methods. The state-of-the-art, as well as the commercial availability of the sensors in these three categories, is discussed in Reference L-6. The conclusions of the study point out that "Many methods and many instruments are available for the detection and location of leaks. No one instrument or method is ideal for any particular application in the space industry". It is further stated that: "It appears that further research is in order to enhance the state of the art in the leak detection area. The ideal system is yet to come".

It appears that incorporating a leak detection system in a space serviceable payload would be a very complicated and difficult task due to the lack of availability of a flight qualified leak detection system capable of detecting leakage of a wide range of liquids and gases.

To date, no fluid couplings exist which can be mated and demated in space and still provide zero leakage service over the expected wide range of fluids and operating environments. There are available some quick disconnect couplings such as a coupling assembly used for RP-1 rocket fuel transfer, manufactured by Snap-Tite, Inc. (Ref. L-3). However, this is intended for ground use, and, as noted in Reference L-6, it is very difficult to assemble. There are also others intended for use with hydraulic fluids or air but none of these are for space use and furthermore there are none which are designed for unamnned operation.

In the design studies in which space-serviceable spacecraft were configured for the DSP, LST, and EOS missons, it was shown that all subsystems requiring fluids could be designed such that they were self-contained within a module. Thus, no fluid interfaces were required. This approach is felt to be a much more reliable, cost-effective, and simpler solution than that of trying to design and develop a fluid coupling capable of in-space connection.

The types of subsystems which require fluid couplings, for example attitude control propulsion, refrigeration systems, etc. are of such a nature that they can be designed to be contained within a single module. If, for example, in the case of a propulsion system, more volume is required for expendables in order to increase the time between expendables replacement or to provide enough expendables for the life of the mission, then one could consider a hybrid system in which extra tanks are mounted on a non-replaceable portion of the payload and connected to the module via plumbing and a joint which could be cut and sealed in a one-time-only operation. After the fuel in the tanks on the module is exhausted, the tanks would be sealed off with a cutter/sealer and the tanks on the spacecraft would be tied into the system through an existing factory-installed plumbing system. The valves, thrusters, and control electronics would be designed with enough reliability to last the life of the payload. In the event of a failure of a valve or thruster or other component on the module which would require module replacement, the module could be replaced with one which would operate only as a self-contained system that would then be replaced at regular intervals as expendables were exhausted. To replace the module would require that the non-replaceable tank on the payload first be sealed off, thus breaking the connection between the module and the spacecraft mounted tank. The remaining fuel in the payload tank would no longer be available for use.

D. REPLACEABLE MODULE/PAYLOAD ELECTRICAL INTERFACE

Design studies in which the DSP, EOS, and the LST were configured for on-orbit servicing via a remote service module have utilized an electrical interface consisting of a single 1.5 in. diameter pin. This pin, shown in Figure L-1, serves three purposes: it provides mechanical alignment between the module and the spacecraft and carries flight loads, it transmits data and commands through a single coaxial connector in the center of the pin, and it provides a DC power and ground connection to each module via the two terminals on the shoulder of the pin.

While the design studies performed and reported used this single pin concept, it is not immediately obvious that a single pin connector is the most advantageous design to utilize for the module/payload electrical interface.

Ground-refurbishable or expendable spacecraft which have no replaceable modules would not need to utilize an electrical interface concept such as the one described herein. Conventional flight-quality, multi-pin, separable connectors could be used as is common practice on existing expendable spacecraft design.

Design studies of payloads configured for on-orbit servicing resulted in definition of an electrical interface between the module and the spacecraft which might be typical of that required for the in-orbit maintainable type of payload. This interface is shown in Figure L-1. The primary feature of the module/spacecraft interface shown in the figure is that it relies on only one mechanical motion to lock the module to the spacecraft which also establishes the electrical and data interface.

The concept of using a data bus for data transmission allowed the use of a single coaxial connector for data transmission. Because the pin had to be large enough to help carry flight loads, it was also large enough to carry electrical power transmission wires (separate DC power and ground wires) in addition to the coaxial cable for data transmission. However, to accomplish data transmission to and from a module via a single coaxial connector required a considerable amount of data bus equipment in each module. In the study reported on in Section 3 of Volume III, no tradeoff studies were performed in which overall payload complexity, complexity of the module/spacecraft interfaces, and complexity of the data bus were compared with the number of electrical connections chosen for data transmission. The number of electrical connections normally made in an expendable payload would bound one end of a tradeoff study and the single pin concept used in the SSDSP, EOS, and LST studies would bound

the other end. It may be possible to reduce the complexity of the data bus system for those payloads already studied by increasing the number of pin connections at the module/spacecraft interface. In addition, the number of pins is a function of the quantity of modules the payload contains, the manner in which the subsystems are divided between these modules, and the amount of redundancy required.

Assuming the number of electrical connections at the interface are increased over the single connector design, it is conceivable that the same single connector concept could be designed to house more than one coaxial connector merely by changing its diameter or its shape; possible pin configurations are shown in Figure L-3. These concepts still allow use of a single pin to locate, align, and lock a module in place, in addition to providing electrical power all with movement of a single pin only. However, these concepts contain more than one coaxial connector for data transmission.

Addition of another pin would still allow the same type of locking motion, actuated by the service module mechanism, to lock the module in place and to provide the power and data transmission. However, the problem of providing close tolerances on the dimensions between the pin holes and the pins from module-to-module and payload-to-payload is introduced. This mismatch in spacing between pins could result in poor contact between the module and spacecraft, misalignment of the module relative to the spacecraft, excess strain on the mating parts, or at the worst, result in failure to achieve a mate. In any case, a dimensional tolerance problem is introduced with the addition of an extra pin which doesn't exist with a single pin.

Another possibility would be the use of a separate multi-pin connector, contained in the module, which would be mated to the spacecraft after mating of the module to the spacecraft. This would require a separate mechanism to achieve connector mate. This mechanism would

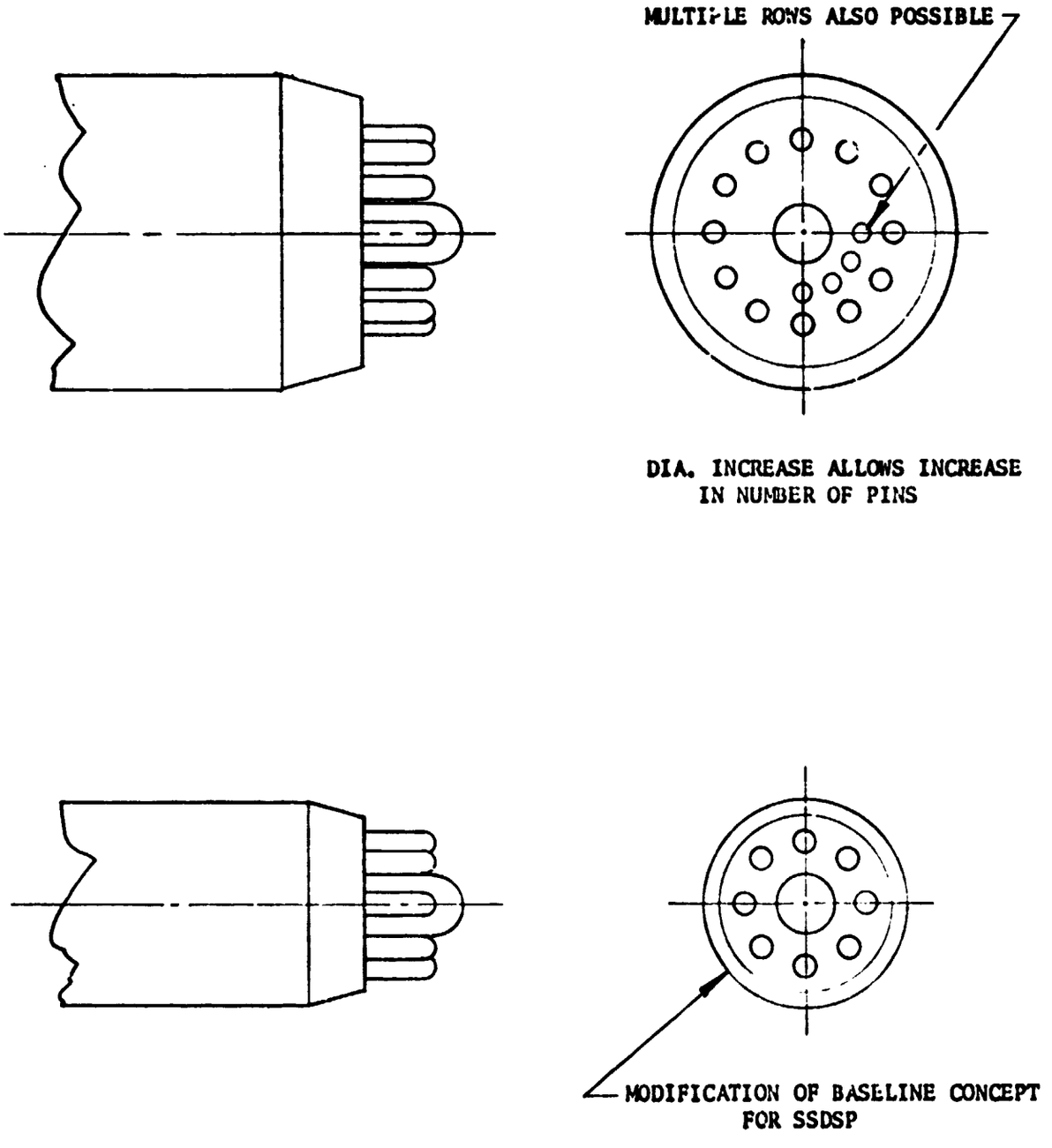


Figure L-3. Modification of Single Pin to Provide for Multiple Electrical Contacts

have to be operated remotely from the service module since the payload is shutdown during the servicing operation. A design could be developed, but at the expense of additional complication, weight and a reduction in reliability.

In each of the above discussions, use of a sliding type of connector for data transmission has been assumed. This is necessary in order to achieve a clean contact. The sliding action of the pin in its receptacle produces a self-cleaning action to assure clean contact. Butt-type or button contacts maintained by spring action are not satisfactory for data transmission due to the possibility of contamination on the contacts. Unlike electrical power transmission, there is not enough current transmitted across the data contact to cause arcing and thus self-cleaning. Therefore, a sliding type of data connector is required.

E. NON-REPLACEABLE EQUIPMENT IN AN ON-ORBIT MAINTAINABLE PAYLOAD

In designing a payload for on-orbit servicing, it is not necessarily desirable to require that all subsystems or components be replaceable. Consideration of the mission life, orbital parameters, type of sensors and equipment, operating requirements of the various subsystems, and packaging requirements for the payload under study may indicate that certain of the subsystems should be designed for the life of the mission and therefore are not to be replaceable.

The most obvious example is the basic spacecraft structure which should be designed for non-replacement. There are no wear out items in the structure and it can easily be designed to withstand the flight loads. The largest flight load factor for the Shuttle is 3.3 g's and is 6 to 7.5 g's for expendable booster load factors. These load factors do not impose severe loading conditions on the structure.

Solar arrays are another item which should be considered for permanent installation. For a typical serviceable payload mission life

of ten years, it has been shown that solar arrays for spacecraft in orbits ranging from low-earth to geosynchronous (Ref. L-1) can be sized to operate satisfactorily for the entire ten-year period. From a design standpoint, increased flexibility can be gained by using a solar array which does not have to be replaced. The array no longer has to be packaged within the module itself which permits greater latitude in selection of the type of array to be used, i.e., roll out, rigid panel, etc. Replacement of the solar array drives, if an oriented array is used, can still be accomplished with a non-replaceable array by means of a technique such as that shown in Figure L-4.

Antennas quite often require special viewing angles or are so large that they present design problems if they are required to be replaced. Consideration should be given to mounting the antenna to the spacecraft structure, and mounting only the associated electronics within the module. The problems which may arise with this approach include the following:

1. If the antenna is steerable, then the drives will probably have to be replaced; methods for replacing drives, but not the antenna, would have to be developed.
2. High operating frequencies (in the RF range) require waveguide connections between the antenna feeds and electronics. If the operating frequency is high (for example, 60 GHz), then the pointing accuracy requirements may place such strict anti-backlash requirements on the drive as to preclude the possibility of designing a replaceable drive. Two problems could arise if the antenna is not replaceable and the high frequency electronics are in the module:
 a. The waveguides must have a separable joint which must be mated to high tolerances (on the order of ± 0.0005 in. for a 60 GHz operating frequency).
 b. Power losses increase as the length of the waveguide increases. This may require that the electronics be mounted directly on the back of the antenna and, if the antenna is small enough, packaging of the entire assembly as a module for removal/replacement.

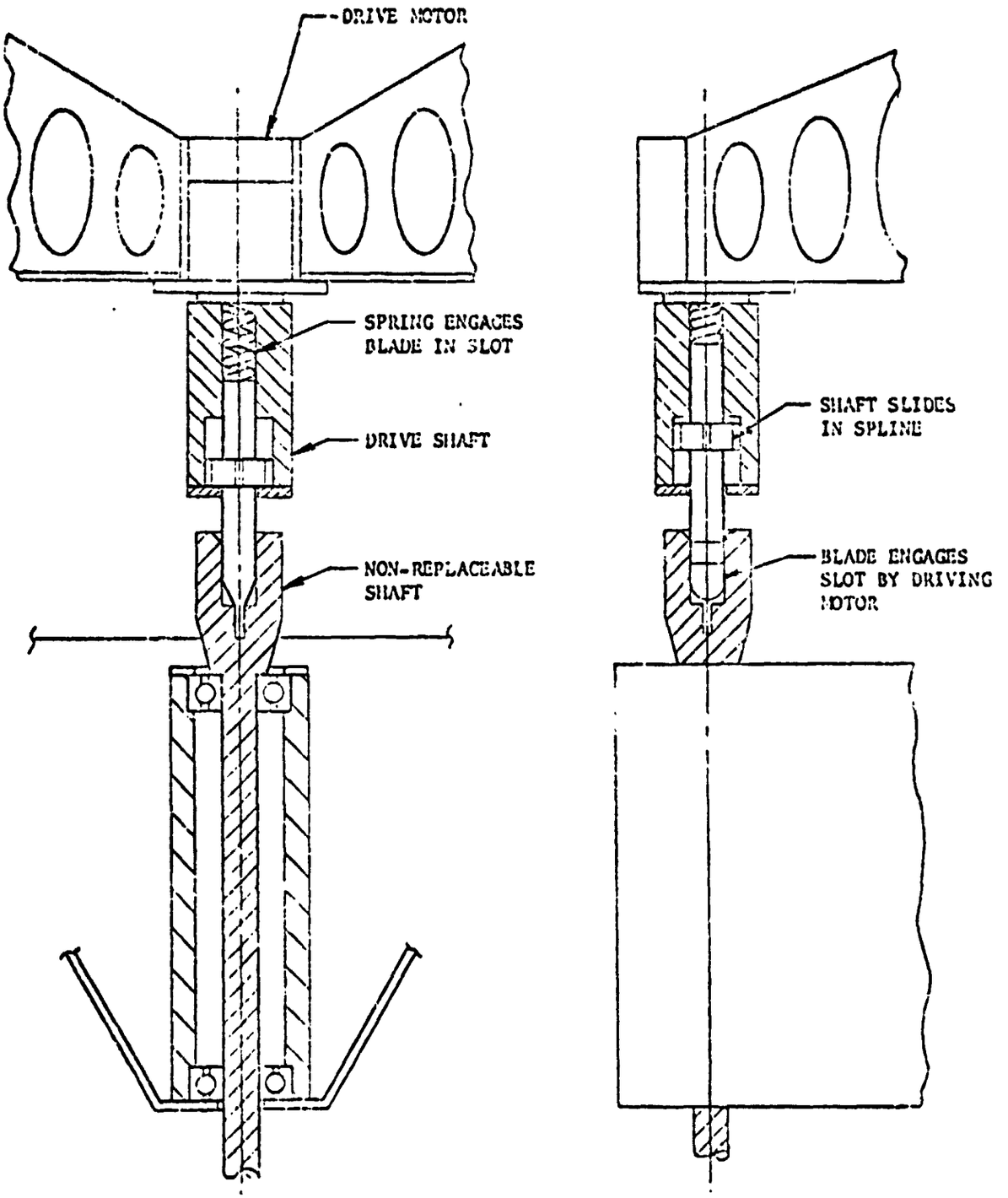


Figure L-4. Replaceable Drive

Optical systems are another candidate for non-replacement. Quite often the optics are very large, as is the case with the LST and DSP satelites. With the exception of drives and focal plane detectors, optical systems can be designed to last for the life of the mission. However, other problems could be introduced. For example, if focal plane detectors are subject to degradation or failure and must be replaced, then some type of focus detection and adjustment system would be required to orient the new focal plane detector.

F. MODULE REMOVAL/REPLACEMENT USING MANIPULATOR

One of the tasks of the operator of the Shuttle attached manipulator system (AMS) is to effect the removal of a module of an on-orbit maintainable satellite and to replace this (failed) module with a new module of the same type by means of the manipulator. This task may be accomplished more simply and effectively if the module system (the module and its grasping points, handles, latches, etc.) is designed to minimize the number of different movements required of the AMS and its operator.

It is not anticipated that the withdrawal/replacement of a module will represent a simple task. Any design features that can be incorporated into the system that will facilitate the operation will simplify the task of the manipulator operator and speed the satellite refurbishment cycle. This will be particularly important in the case of a payload containing a large number of modules where equipment is being replaced down to the "black box" level.

The time required to repair/refurbish the module complement of a low-earth-orbit, on-orbit maintainable satellite can be reduced if the modules are designed so as to minimize the number of motions required of the attached manipulator system.

A design analysis that is presented in Section 3 of Volume III was made in the form of a preliminary design (layout drawing) to identify a

feasible means of designing a space replaceable module, i.e., satellite module that could be readily removed and replaced by means of the operator-controlled attached manipulator system.

Several perturbations were made to the drawing. One design (Figure L-5) was made wherein the latch handles and module carrying point were designed in the form of simple rods. The latch release handles were located symmetrically on either side of the central grasping (carrying) point. In order to remove a module, this design would require the manipulator to first release the first (one) latch via a grasp, release, and pull operation. Then it would be necessary to translate the end effector of the manipulator to the opposite side of the front face of the module and repeat the above operation at the second latch. Finally, a translation could then be required back to the central grasping (carrying) point to grasp the module handle and withdraw the module from its structural housing. A complete reversal of these operations would be required to insert and latch the replacement module.

Further examination of the design led to the identification of a means whereby latch release (or engagement) and module extraction (or insertion) can be accomplished at one single grasping point thus eliminating the numerous translational motions previously described.

This simplification can be effected (Figure L-6) by designing the module grasping point as an integral part of a slide assembly which is a mechanism containing a ball-lock-type detent or spring-loaded plunger that is seated in a suitably shaped groove. The end of the slide opposite the grasping handle is attached to a toggle arrangement that directs (electrical) power via switches to linear actuators controlling closing/opening of the module latches. Movement of the slide from one detent to another opens (or closes) the latches. Continuation of the movement (for example, a "pull" to unlatch) results in extraction of the module. With this design, only one straightline motion (no rotations or lateral translations) at one point is required to extract or replace a module.

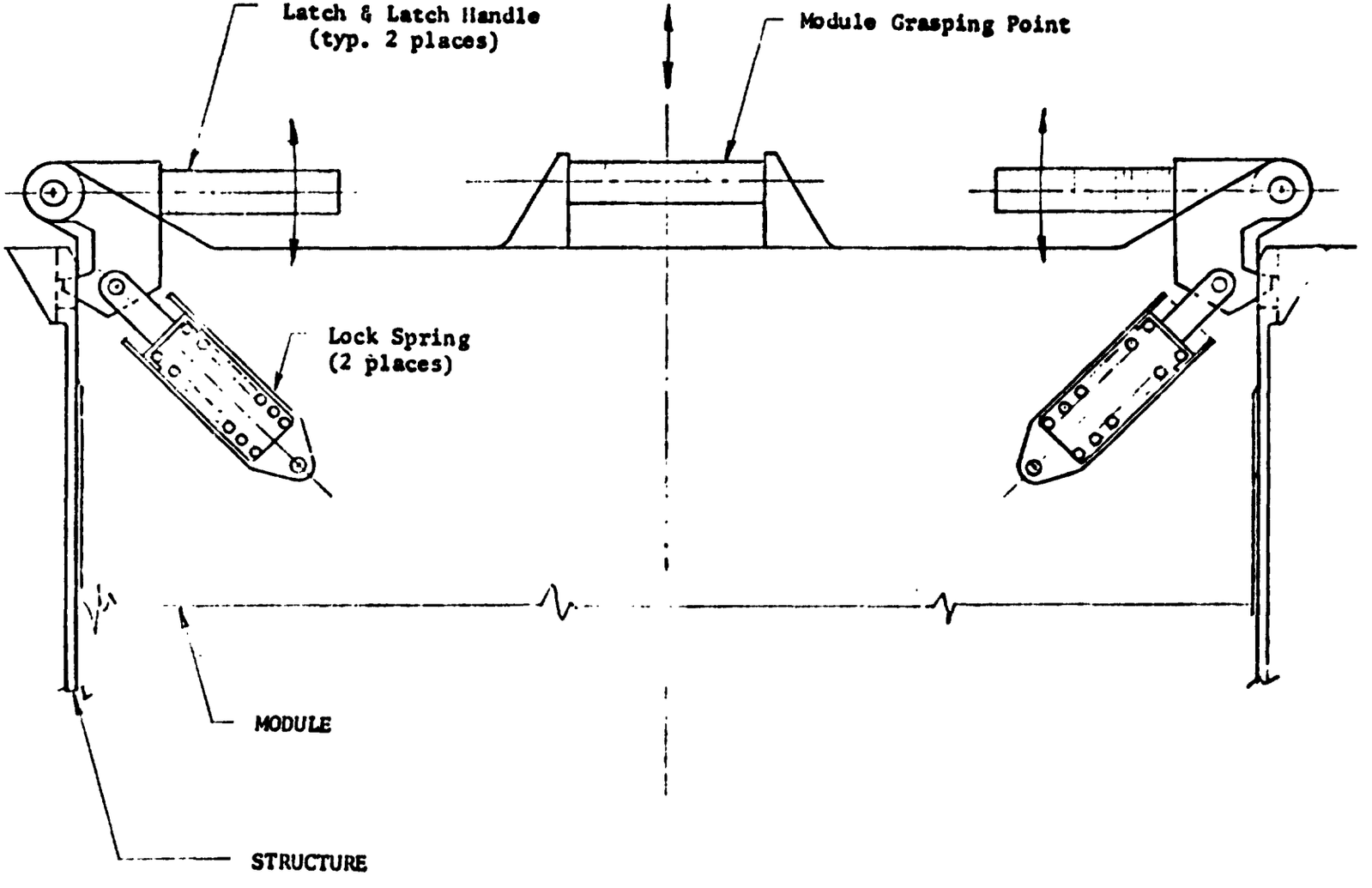


Figure L-5. Module Handling System, Concept 1

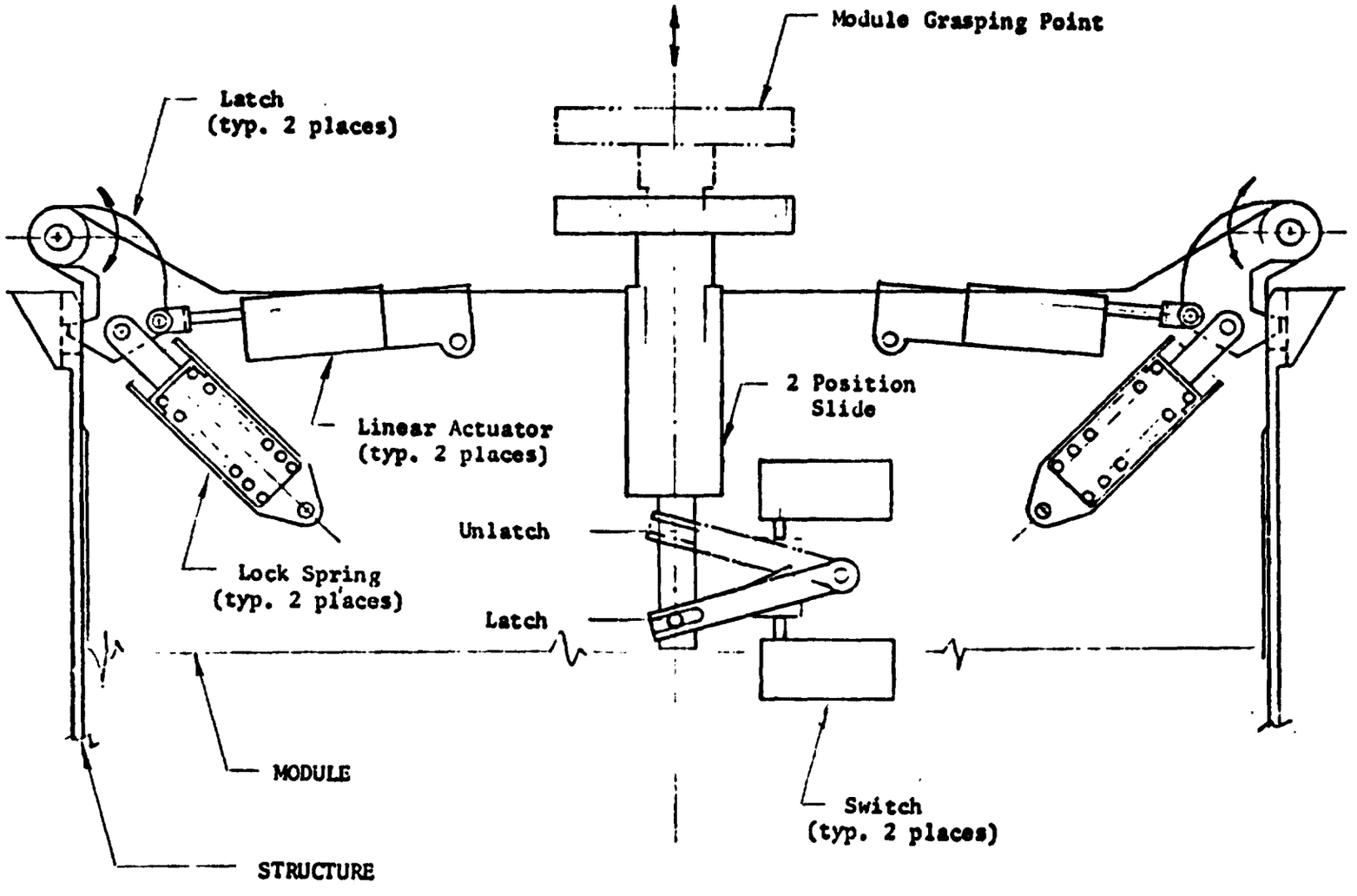


Figure L-6. Module Handling System, Concept 2

It is recommended that the modular elements of on-orbit maintainable payloads be designed to reduce the number of motions and the number of points at which the required motions are carried out to a minimum in order to facilitate module withdrawal/inspection and minimize the time in which this function may be carried out.

G. MODULE BACKUP HANDLING PROVISIONS

A primary module handling system is required for satellite module replacement by the orbiter attached manipulator system. This system would consist of the latches, grasping points, handles, etc. required to transfer modules between orbiter-carried spares and the attached satellite via employment of the manipulator system. A backup module handling system would ensure completion of this task in the event of primary system failure.

The cost effectiveness of the Shuttle depends to a great extent on the ability to repair and/or refurbish the modular components of low-earth orbit satellites. A secondary or backup mode of module replacement would permit the successful accomplishment of this task in the event of primary module handling system functional failure. A supplementary backup system can be provided with little or no addition of hardware elements to the basic system. The versatility of this system can be enhanced by the addition of a backup system that would be operable by the manipulator if a failure occurs in any of the elements (mechanical or electrical) comprising the primary system.

The primary system described as an automatic system is designed to minimize the number of operations required to effect module change-out by means of the manipulator. The system contains electrically controlled linear actuators that are used to operate the latches securing the module(s) to the satellite structure. If the actuators were equipped with an over-ride device (e.g., a slip clutch) and the latches designed to

furnish integral external handles (grasping points), as shown in Figure L-7, the latches could always be released and/or engaged by the remote manipulator thus making it always possible to mechanically replace a module in the event of a mechanical or electrical failure in any element of the primary module handling system.

H. RECOMMENDED TRADEOFF STUDIES

1. DOCKING SYSTEMS

For the design studies performed as an aid in developing design guidelines, a modified Apollo probe and drogue docking system was selected for use with the service vehicle concept. The system was modified for unmanned use by removing those components normally used for the manned removal of the probe from the Apollo crew transfer tunnel. In addition, an indexing latching system, consisting of three latches on the nose of the probe which would engage in three slots in the bottom of the cone once the service module was indexed property relative to the spacecraft, was incorporated.

The Apollo docking system was selected for the design studies because it is a proven workable system and because it was easily adaptable to the concept generated for the space serviceable spacecraft. However, no tradeoff studies were performed to determine the impact of the type of docking system chosen on the design of the spacecraft, service module, replaceable module, or module transfer techniques.

Other docking systems have been studied and designs generated. For example, described in Reference L-7 is a docking frame with capture latches and guide arms which may offer some advantages over the Apollo system, depending on the type of spacecraft configuration chosen.

2. SINGLE VS. MULTIPLE LAUNCHES

Tradeoff studies should be performed to determine if there is any advantage in designing the spacecraft (ground refurbishable and expendable

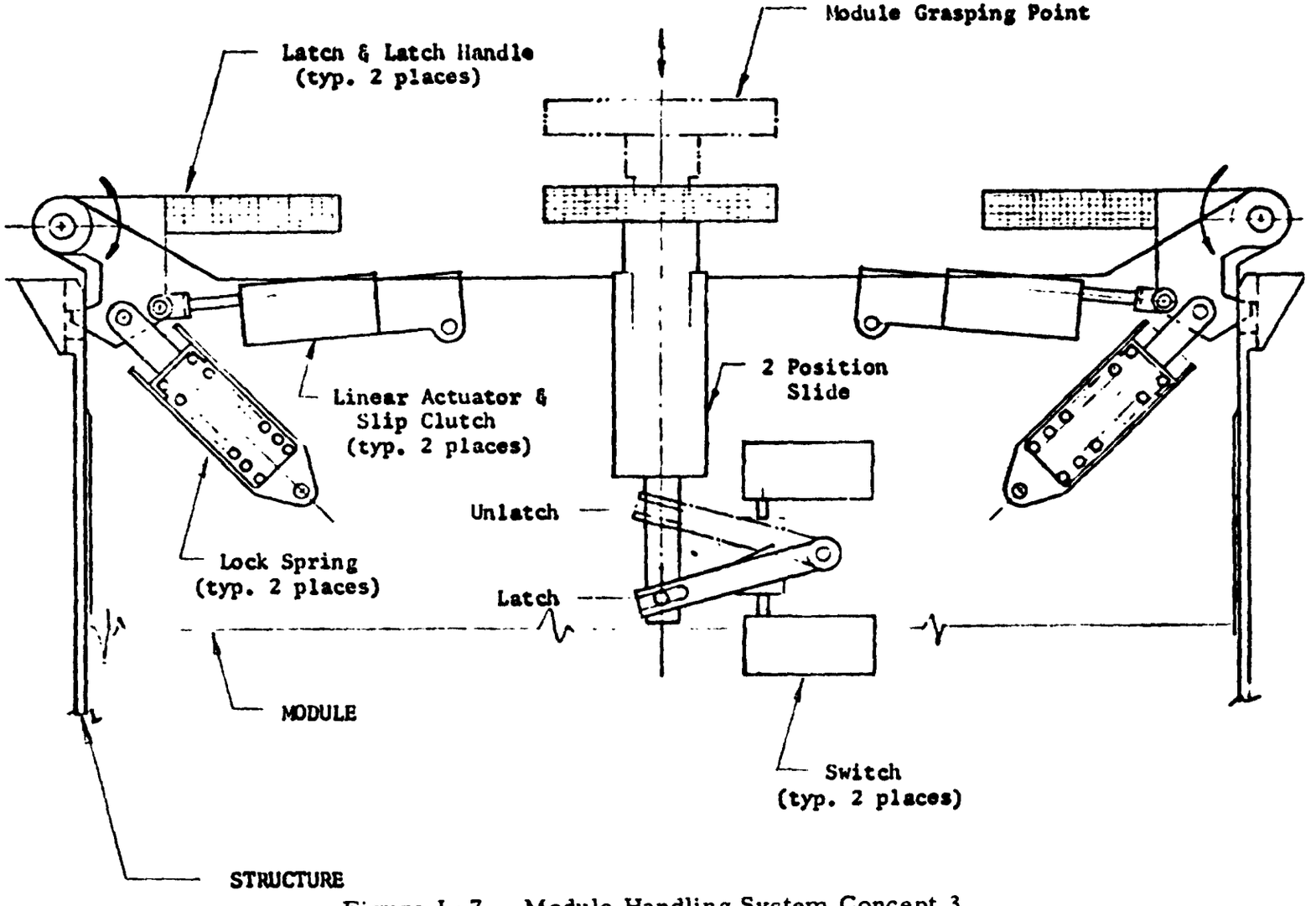


Figure L-7. Module Handling System, Concept 3

as well as on-orbit maintainable) such that more than one can be launched with the Shuttle.

Some spacecraft operational schemes will probably require more than one payload of the same type spaced apart within the same orbit or perhaps in different orbits. To satisfy requirements of this nature, it could be advantageous to launch more than one payload with a single Shuttle. The weight and geometry of the given payload must, of course, be such that the Shuttle payload bay can accommodate more than one payload. Assuming this requirement is satisfied, the impacts on the design of spacecraft for multiple launch should be determined. This would include determination of extra attach structure required, increased loads on the basic structure, release mechanisms, and associated failure modes, etc.

Multiple launches should not be limited to payloads of the same design. It may also be advantageous to consider launch of two completely different types with a single Shuttle launch.

I. REFERENCES

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L-7. DOD Upperstage/Shuttle System Preliminary Requirements Study, Vol. 3, McDonnell Douglas Astronautics Co., Huntington Beach, Calif. (August 1972).

APPENDIX M. RELIABILITY

To achieve cost savings with on-orbit repair, failure modes leading to loss of payload or preventing on-orbit repair must be identified and their effect minimized. Addition of selective redundancy to critical areas or simple design changes are generally all that is required. To insure repair success and reduce revisit frequency, sufficient telemetry and instrumentation are required to identify the failed module and other modules with such extensive degradation that they will likely fail before the next scheduled revisit. The ability to retrieve failed modules from orbit will allow realistic corrective action to be taken on future payload designs.

A. EXPENDABLE PAYLOADS

Although modularization is not inherently required for expendable payloads, it can lead to early standardization and to decrease test problems. For example, if three different payloads each require a 12-volt regulator all with the same requirements except current output, a separate regulator would be built for each program. It might be desirable to use the one with the maximum current output for all three, but this is usually prevented by weight, size, configuration, and interface restrictions. In the Shuttle era, a modularized standard regulator could be used in all cases, since these restrictions can be removed by standardization.

Curve A of Figure M-1 is a typical reliability growth curve using today's methods, while Curve B is a hypothetical growth curve for a similar program in the Shuttle era. The large improvement in reliability should result from usage of standardized modules. Some of these modules will already have had considerable reliability growth, since they may have been used in a variety of previous program. Table M-1 lists some of the advantages of standardization and how they affect the program reliability, design, cost, and schedule. High reliability (hi-rel) parts will probably still be required for the same reasons as today.

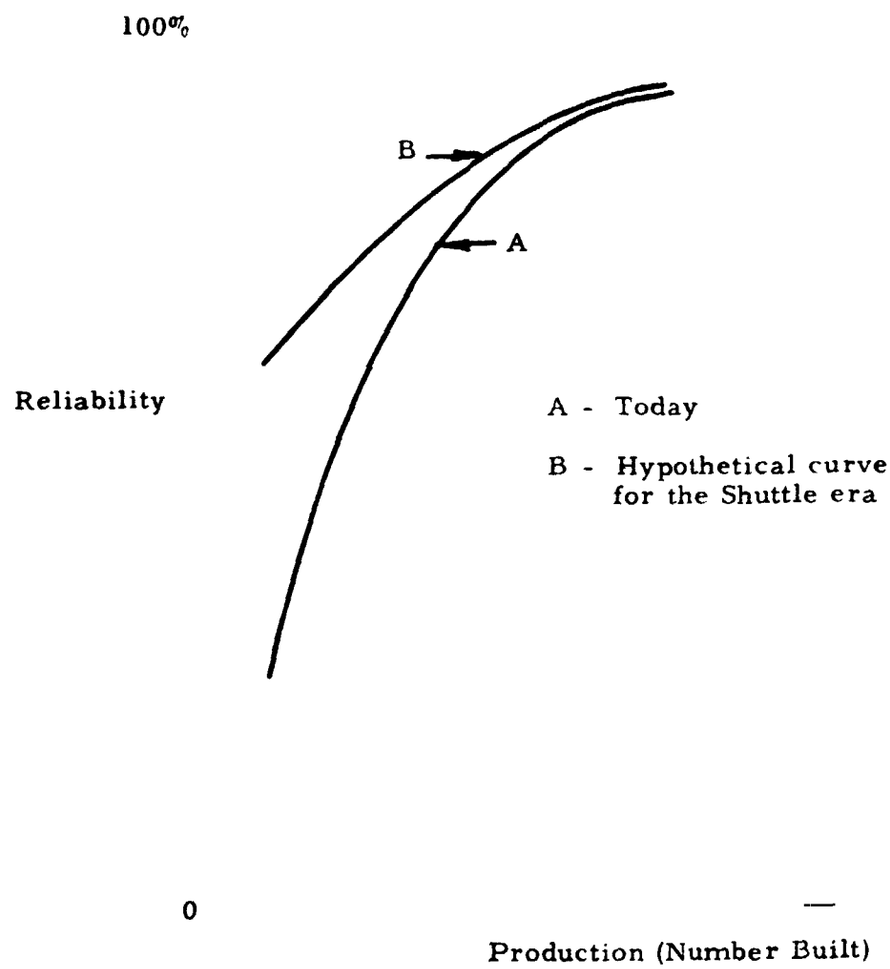


Figure M-1. Reliability Growth of a Typical Payload

Table M-1. Advantages of Standardization

No.	Advantage	Main Effects			
		Improve Reliab.	Improve Design	Reduce Cost	Reduce Repair Time
1	Electrical Design	X	X	X	
2	Manufacturing	X		X	
3	Testing	X		X	X
4	Inspection Procedure	X		X	
5	More data on weak areas	X	X		
6	More reliability data	X	X		
7	Improved TLM	X	X	X	X
8	Higher availability of spares			X	X
9	Fewer errors in servicing	X		X	X
10	More complete servicing	X		X	X
11	Improved test equipment availability			X	X
12	Improved jig and fixture availability			X	X

B. ON-ORBIT MAINTAINABLE

For on-orbit maintainable payloads, most failures are corrected by replacement of modules on orbit. The cost of on-orbit repair includes the cost of the module to be repaired plus the shared cost of the Shuttle flight. If ground repair is required, the cost includes the repair cost plus the cost to bring the payload to earth and after repair, to replace it in orbit. If a spare payload is available on the ground, it can be used to replace the failed payload (on orbit). Cost savings should result, since the failed payload can be returned to earth on the same trip. If a payload is "lost" and cannot be retrieved, a new payload must be built and placed in orbit. An approximate cost and cost ratios for various strategies are shown below for a $30M unit cost payload and $10.3M launch cost:

Strategy	Launch Cost	Repair Cost	Total Cost	Approx. Cost Ratio
On-orbit Repair	$10.3M/4	$ 2M	$ 4.58M	1.0
Ground Repair	$10.3M/2	$ 2M	$ 7.15M	1.6
Payload Replacement	$10.3M	$30M	$40.3M	8.8

The cost of an on-orbit maintainable payload is greatly reduced, e.g., from $40.3M to $4.58M; however, some change should be expected in the part quality requirements area. If the choice were between hi-rel and MIL STD parts, then hi-rel parts would still generally be selected since MIL STD parts failure rates are about five to ten times higher than high reliability parts. The MIL STD parts are two to three times lower in cost, but parts represent only 10% of the payload unit cost. Using the previous example, the payload unit cost will be less for MIL STD parts by:

$$\Delta \text{ parts cost} = (\$30M)\ (10\%)\ (3-1) = \$2M$$

However, the operating cost will increase due to the increase in payload replacement rate. As an example, if a five times increase in parts failure

increases payload failures by two times to ac ount for nominal redundancy, the operations cost will increase by:

$$\Delta \text{ operations cost} = (\$4.58\text{M})\ (2\text{-}1) = \$4.58\text{M}$$

or a total increase of $2.58M ($4.58M - $2.0M) using MIL STD parts. The reliability of a MIL STD unit can be increased by increasing redundancy to reduce the payload failures. This approach must be traded against increases in cost for parts, testing, and refurbishment. Rather than using MIL STD parts which have high failure rates, intermediate quality parts may be more appropriate, since high reliability parts are basically established from the rigorous screening test of MIL STD parts.

Figure M-2 shows the effect of failure rate from putting more money into screening. Table M-2 shows these different screening requirements, while Table M-3 shows relative cost, failure rates, and lead times for the devices. Although data showing the effectiveness of individual screening tests is generally not available, it is suspected that certain tests have not been optimized for screening effectiveness.

1. Test Time - The burn-in time for Class B parts (Table M-2) is 168 hours (one week). It may not have been optimized.
2. Obsolescence - When centrifuge testing was first performed (on transistors in the 1950's), it was an effective method of checking poor lead bonding. However, today, integrated circuit leads are much lighter, and this test rarely generates enough force to show a weak bond (that would not show in another test).

Although screening tests will be maintained, reduction may be advisable in certain areas. The criteria generally would be that any test, whose cost was not less than the savings resulting from the decrease in repair flights costs (because of failure rate reduction), should not be incorporated.* In some cases, redundancy may be more cost-effective than adding a screening test, although this may lead to more extensive on-orbit repairs.

*All costs connected with the higher failure rate must be considered; e.g., additional ground testing and repair, more extensive on-orbit repair, and longer periods of outage.

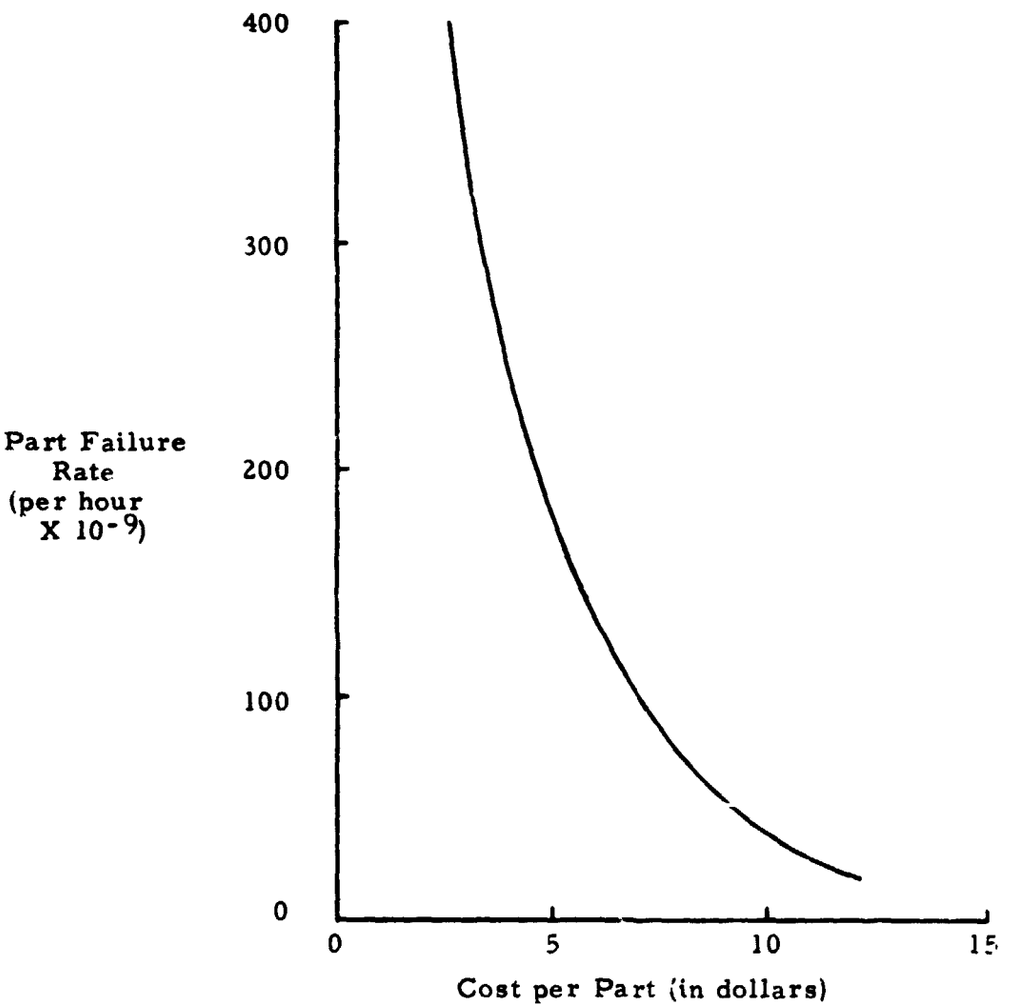


Figure M-2. Reduction in Failure Rate Resulting From Addition of Screening Tests (IC's per MIL-STD-883)

Table M-2. Screening per Method 5004-MIL-STD-883

TEST	CLASS A	CLASS B	CLASS C
Precap Visual	Yes	Yes	Yes
Bake (Stab.)	Yes	Yes	Yes
Thermal Shock	Yes	No	No
Temperature Cycling	Yes	Yes	Yes
Mechanical Shock	Yes	No	No
Centrifuge	Yes (High)	Yes (Medium)	Yes (Low)
Hermeticity	Yes	Yes	Yes
Electrical	Yes	Yes (Less)	No
Burn-In	240 hr	168 hr	No
Reverse Bias (burn-in)	Yes	No	No
Radiograph	Yes	No	No

Table M-3. Failure and Cost Information on MIL-M-38510 IC's (T^2L-Construction)

	CLASS A	CLASS B	CLASS C
Failures per 10^{-9} hr	20.4	40.7	107.9
Cost* per Device (in Dollars)	12	8.5	2.5
Procurement Lead Time	20 weeks	4 weeks	In Stock

* For procurement lots at 1000 devices.

One area in which MIL STD parts might be considered is in the experiements under the following conditions:

1. Data are not urgently required.
2. The experiment is a small part of the mission.
3. Many other more important experiments are carried.
4. Failure of the experiment doesn't affect other areas of the payload.

Then, if one of this type experiment fails, it will not be repaired until the payload is visited for some other reason.

Today, if a payload fails, the only information on the cause of failure comes from telemetry (TLM) data. Often this is not effective in really explaining what has happened. Because of this:

1. Extra design time and effort are used before flight.
2. Extra testing is performed before flight.
3. Extra TLM and instrumentation are required even in non-critical areas in order to pinpoint causes of failure.
4. When failures actually occur, extensive and expensive test and design evaluation programs are conducted (on the ground).
5. Since the problem still cannot be pinpointed in some cases; e.g., low frequency, intermittent-type failures, extra redundancy and other expensive and cautious design approaches are used on succeeding flights.

In the Shuttle era, the failed module can be returned to ground and a complete failure analysis performed. Degraded modules that were also being changed, together with experimental data, could also be returned and analyzed.

The general concept of degraded or payload survival state is important in preventing payload loss. Although requirements vary from payload to payload, this concept of degraded state is illustrated for the following subsystems.

1. Attitude Control - For many experiments very fine control of attitude is required to provide proper pointing, platform stability and position determination. However, to prevent tumbling and insure safe pickup a less complex attitude

control system can be used. This can be accomplished either by using a low cost completely redundant attitude control system or by using selective redundancy in the present system. For example, a lower gain feedback amplifier might be used as backup for the high gain amplifier now required for high stability.

2. Thermal Control - Thermal control is required for stability of operation and to prevent damage from excess heat or cold. Module thermal independence is generally desirable. After failure, while awaiting payload repair, it is mainly required to supply sufficient heat to prevent damage by cold. Thus, to achieve this objective one change might be to always use two smaller heaters, rather than one large one, so that either could supply enough heat to prevent widespread damage in a module, if the other heater failed.

3. Partial Operation of the Computer - During normal operation the computer is the brains of the mission. After failure it is still required to provide control of functions such as attitude control, thermal control, and the power distribution (to critical areas). The computer could either be made partially redundant in these areas, or be programmed, after failure, to use other areas that usually control non-critical functions.

4. Communications - After failure, sufficient up and down link capacity must be maintained to insure operation of those housekeeping functions, required for survival, which are controlled from the ground. They will vary considerably from payload to payload, i.e., the greater the on-board computer capacity the less the requirement. An extreme example of high requirements would be a "fly by wire" mission. Usually less control is required and a degraded system capable of detecting safety related items, e.g., leaking fuel, and providing some backup for the on-board computer would be sufficient.

5. Power - The payload design should be such that a lower level of both regulated and unregulated power necessary to insure survival is provided after single point failures of the power system. The design of the experimental area should be such that, except for thermal control, it does not require power for survival while awaiting repair.

Table M-4 shows how a single string power supply might be changed to prevent loss of the payload after a failure. Appendix N shows how the present FMEA procedures could be extended to verify this type of effort had been accomplished throughout the payload. In this example, it is

Table M-4. Example - FMEA of a Single String Power Supply

COMPONENT	MINIMUM PERFORMANCE REQUIREMENT*	EFFECTS OF FAILURE	RECOMMENDATIONS
Battery and Circuitry	1/2 Power	Loss of Payload	Split Battery Into two Parts**
Solar Array and Circuitry	1/2 Power	None, if Repaired Quickly	Respond to Failure Within Two Weeks**
Regulators and Distribution Circuitry	1/5 Power	Loss of Payload	Separate Regulation For Main Users Make Critical Regulators Redundant

* To prevent loss of payload (this is supplied by engineering).

** All but critical loads must be removed if these failures occur.

assumed that one-half of normal power is required for payload survival including one-fifth of all regulated power. This power is used in critical areas of the subsystems. The requirements of power necessary to prevent loss of the payload are the minimum power requirements for the attitude control, thermal system, command, and communications to insure that the payload survives until retrieved.

This example shows how design changes, operational change, and selective redundancy can be used to prevent loss of a payload. For a design change, the single battery would be replaced by two batteries, each half the size of the original battery, so that either could supply power for survival after the other had failed. After failure, all non-critical items would be turned off and on-orbit repair would be scheduled before the batteries became depleted. An example of selective redundancy is shown in the voltage regulator area. Each regulator is divided into two sections, one for critical and one for non-critical users. The critical section would be made redundant. It should be noted that program cost savings result from these changes.

If the module that contains the failure is totally unknown, the payload must be returned to the ground for refurbishment. If the failure can be isolated to a small number of modules, replacement modules can be carried in the Shuttle and changed on the payload.

The impact of failure identification on the required number of launches per year is shown in Figure M-3. The top set of curves shows the reliability functions of the various modules, including a truncated module. The system reliability is determined by multiplying all the module reliabilities together.

The bottom set of curves is a plot of number of launches per year against the number of modules changed during each on-orbit repair visit including the failed module. The curves range from no information, except that a specific module has failed, to some information on the status of other modules. If perfect information were available, the launch rate would be even lower.

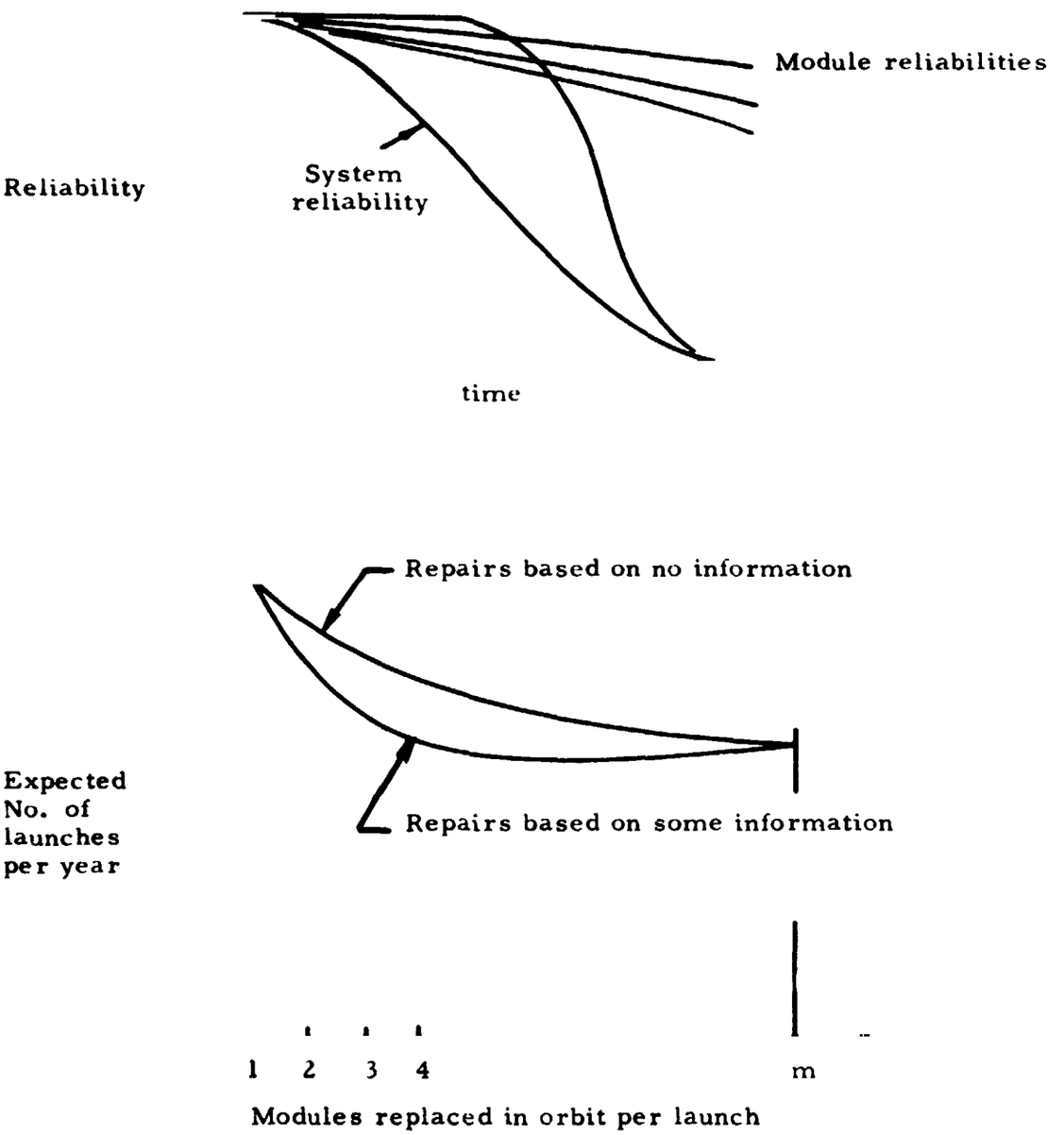


Figure M-3. Impact of Failure Identification

A few of the indicators of module status that may foretell early failure are:

1. Redundancy - The module failure rate will increase after redundant circuit(s) in a module fail.
2. High Temperature - A module operating above its predicted temperature may be approaching failure
3. Experiments - Immediate repair of a failed experiment may not be necessary; however, it would be repaired the next time the payload was visited for some other reason.
4. Wearout - Several components give signals indicating the approach of wearout. These include batteries giving poor voltage recovery after discharge during an eclipse, TWT's with low emission, and electromechanical devices indicating noise and sluggish operation.
5. Fuel Depletion - The fuel reserve is usually measured today and should be continued on these types of payloads.

Using standardized modules on this type of payload is an advantage, since almost all failure modes that lead to loss of payload or prevent on-orbit repair will have been identified and appropriate corrective action will have been taken through maturity. Also, the TLM-instrumentation will be developed to monitor module degradation information.

C. GROUND REFURBISHABLE

Much of the above analysis holds true for ground-refurbishable payloads; however, the economic advantages of eliminating failure modes which can cause loss of the payload are expected to be smaller. Detailed TLM and instrumentation are not required for this type of payload. They need only be sufficient to show that the payload is safe for retrieval, to show when major failure modes are approaching (on-orbit), and to aid in trouble-shooting during ground repair.

Standardization is important for the same reasons as with on-orbit repair type payloads. Also, modularization will allow design improvement to be added during refurbishment. Only rarely will other than hi-rel parts be cost-effective, e.g., elimination of very expensive screening tests that

provide little reduction in failure rate. Payload can be returned from orbit and failure analysis performed in the same manner as with on-orbit maintainable type payloads.

D. SORTIE

In Sortie flights, all failures result in loss of the mission except for those that can be repaired during flight. The only requirements are that failures do not affect Shuttle housekeeping equipment and the crew environment must be preserved. This mission type has low TLM and instrumentation requirements. Instead, test points may be available for use by crew members (using hand instrumentation). Since many payload (experiments) use the same function repeatedly, e.g., amplifier in a solid-state antenna, a few of these could be carried and repairs made by the crew, if necessary. This represents another reason for modularization, standardization, and interchangeability requirements on this type of mission. MIL STD parts can be used in areas where repairs can be made on orbit. Not only can failed modules be returned to earth, but tests can be conducted on orbit for even better failure information.

E. CONCLUSIONS

System engineering must consider major failure modes so that requirements can be placed on the subsystems during early design stages to minimize the effects of such failure modes.

System engineering must specify the TLM instrumentation requirements necessary to identify failed modules and major degradation of modules.

The subsystem designers should consider approaches that will use features provided by the Shuttle, e.g., splitting the battery into two parts and low level redundancy in critical areas.

A method must be developed to insure that the above tasks have been satisfactorily accomplished. Present FMEA procedures should be extended to cover these tasks.

Standardized modules should be used, where possible, for all housekeeping payload functions.

The effectiveness of individual part screening tests should be reviewed to determine their effectiveness in reducing payload costs.

Redundancy should be considered in lieu of expensive screening tests, expecially on on-orbit repairable and sortie payloads.

The ability to perform failure analyses on hardware returned from space will decrease design complexity and R&D, testing time and cost, and increase reliability on future flights.

F. REFERENCES

M-1. J. J. Gardner, SSD Electronic Parts Orbital Failure Rates (Active and Dormant Operation) and Associated Failure Modes, Report D154080, Lockheed Missile and Space Co., Sunnyvale, Calif. (25 July 1972).

M-2. E. E. Bean, C. E. Bloomquist, Addendun to Reliability Data From In-Flight Spacecraft; 1958-1970, NASA Report PRC D-1864, Planning Research Corp., Los Angeles, Calif. (30 November 1972).

APPENDIX N. FMEA PROCEDURE

This is an example Failure Modes and Effects Analysis (FMEA) to illustrate how the FMEA may be extended to the Shuttle payload.

1. PAYLOAD TYPES

These procedures are intended to be used for on-orbit maintainable and, where applicable, with ground refurbishable payloads.

2. PURPOSES

The purpose of this procedure is to extend the present FMEA procedures so that following additional information can be obtained.

(1) Identify areas where failure(s) lead to the loss of the payload.

(2) Identify areas where failure(s) prevent repair of a payload on orbit.

(3) Identify areas where failure(s) may require more than one re-visit to repair a payload on-orbit.

(4) Identify TLM indicators that will aid in the selection of modules to be changed during on-orbit repair, in addition to the failed module.

(5) Identify areas where new instrumentation and TLM should be considered.

3. ASSUMPTIONS

The following activities must be performed before this procedure can be used.

(1) The payload requirements relative to Shuttle safety must be established.

(2) The requirements for payload survival both at the system and subsystem levels must be established. (See Attachment 1)

(3) The requirements for identification of failed modules must be establi-hed at the system and subsystem level. (See Attachment 2).

(4) An FMEA analysis of the type presently performed for NASA payloads must be accomplished and the following information must be available (for each failure mode).

(a) The effect that the failure mode has at the subsystem level.

(b) The effect that the failure mode has at the system level. (In addition the effect, must be shown, as if redundancy were not present, if applicable.

(c) The TLM or other indication that shows where the failure has occurred.

(d) An identification number.

4. PROCEDURE

4.1 Failure Classification

The following procedure shall be followed to classify failures, reduce the likelihood of serious modes and identify those that remain.

(1) Insure that a FMEA has been performed and the type of information supplied in Attachment 1 and Attachment 2 (if applicable) is available.

(2) Classify each of the failure modes per the table and according to instructions given in Attachment 3.

(3) Determine that all Class 1 (single point safety) failures have been eliminated, if possible, by design change or with redundancy. (It seems likely that a Shuttle safety organization will be required to approve any exceptions).

(4) Recommend corrective action to reduce the effects of Class 2 and Class 3 failures (e.g., selective redundancy). Where the corrective action seems expensive, or otherwise undesirable, a cost analysis shall be performed.

(5) Prepare a list showing all remaining Class 1, 2 and 3 failure and an explanation of why they had not been eliminated.

4.2 Indicator Classification

The failure indicators shall be reviewed (for each failure mode), to insure that they comply with the requirements of Attachment 4. Where deficiencies are noted new instrumentation and TLM channels shall be recommended.

4.3 Module Selection

For each space repairable module, a list shall be prepared cross referencing the failure mode identification number to the failure classes as shown in Attachment per the example in Table I. This list shall be used as a guide for selecting modules to be changed, in addition to the failed module, during on-orbit repair, e.g., a Class 4 or 5 failure has top priority, since either indicates a SPF that could lead to loss of the payload.

Table I Failure Mode Classification Example

Failure Class	Failure Mode Identification Number	
4	7, 9, 11	Example ID Numbers
5	12, 15	
6	10, 9	

APPENDIX N, ATTACHMENT 1

SAMPLE OF PAYLOAD SURVIVAL PROCEDURE

1. PURPOSE

This Attachment is the procedure that would be followed by design engineering to identify conditions required for payload survival.

2. REQUIREMENT

The payload shall be analyzed to determine the minimum type(s) and level(s) of outputs required for its survival until repaired on-orbit or returned to earth, for repair. The information shall be obtained at the subsystem level (or lower if required).

3. DEFINITION

A payload is considered "lost" under the following conditions.

(1) It cannot be recovered or is unsafe to recover.

(2) It has extensive damage and repairs are uneconomical.

4. CONDITION LEADING TO PAYLOAD LOSS

Listed below are conditions that can lead to loss of payload together with the subsystems that can affect these areas.

4.1 Payload Tumbling

The payload can tumble (or be in an unrecoverable orbit). This can result from selective failure of the attitude control, power, computer or communications subsystem.

4.2 Payload Unsafe

The payload may be unsafe to dock with due to failures of docking mechanism or leakage of dangerous substances. They include hydrogen, chemicals or radioactive material. Also the instrumentation - TLM (and power) necessary to determine the status of these dangers must be operational.

4.3 Failure Isolation

A failure may cause secondary failure(s) which make the payload unrecoverable or too expensive to fix. The transient effect of each failure mode must be considered. Anv subsystem has the potential to cause this type of problem, e.g., heat from a short before removal from the line. Also the shock from a violent A/C maneuver, upon failure, must be considered.

4.4 Thermal

After failure, the payload must be able to stand the thermal environment of space. This requires thermal control, heaters, heat shields, power and possibly A/C and communications.

4.5 Capture or Storage Damage

The payload can be lost during capture if the docking mechanism is defective or if sudden failures occur in the A/C and related areas. No problems during stored reentry have been identified.

5. OUTPUTS

The results shall be used as a guide for subsystem designers and FMEA evaluation personnel. The following is typical of the information to be provided on some of the subsystems.

5.1 Power Subsystem

(1) Unregulated power

(2) Regulated power

(3) Time dependence of power

5.2 TLM

(1) Critical channels

5.3 Computer (Controller)

(1) Required action and channels

5.4 Attitude Control

(1) Transients allowable

5.5 Thermal

(1) Minimum heaters

(2) Heater controls

5.6 Communications

(1) Which transmitter and receivers are needed

5.7 Safety

(1) Any safety areas

APPENDIX N, ATTACHMENT 2
SAMPLE OF FAILED MODULE IDENTIFICATION PROCEDURE

1. PURPOSE

This Attachment is a sample of the type of procedure that would be followed by design engineering to identifying conditions preventing on-orbit repair.

2. REQUIREMENT

The payload shall be analyzed to determine the minimum type(s) and level(s) of outputs required to insure identification of the module that has failed. This information shall be obtained at the subsystem level (or lower if required).

3. DEFINITION

A payload is considered unrepairable if any of the following occur.

(1) Communication is lost with the payload

(2) The TLM information is insufficient to isolate the failure to three replaceable modules.

(3) The module can not be changed on-orbit.

4. NON-REPAIRABILITY

The following condition can lead to the payload not being repairable on-orbit (in one trip).

4.1 Communications

Communications necessary to identify the failed channel may be lost. This can be caused by critical failures of TLM channels, communications, power, and certain failures of the A/C and its control which make communications impossible, e.g., the antenna(s) are out of position.

4.2 TLM Information

Insufficient TLM information usually results from limitations on TLM channels and/or instrumentation. Sometimes the failure can be isolated to two (three) replaceable modules and all can be changed but at a cost penalty.

4.3 Module Replacement

The module may not be replaceable (on-orbit) if the exchange mechanism is jammed or if an output of the module, to be replaced, is required during the replacement period.

5. OUTPUT

The results shall be used as a guide for the subsystem designers and FMEA evaluation personnel. The following is typical of the information to be provided on some of the subsystems.

5.1 Power Subsystem

(1) Regulated power

5.2 TLM Channels

(1) Critical channels (e.g., some may be lost and failed module still can be identified).

5.3 Computer

(1) Control in TLM area.

5.4 Instrumentation

(1) Critical areas

5.5 Attitude Control

(1) If applicable to communications

APPENDIX N, ATTACHMENT 3
SAMPLE OF FAILURE CLASSIFICATION PROCEDURE

1. PURPOSE

This procedures provides information to the FMEA evaluation personnel for grading the effect of failures. This information is necessary for design change recommendations and to aid in selection of modules, other than the failed module, to be changed during on-orbit repair.

2. REFERENCES

(1) Attachment 1

(2) Attachment 2

3. PROCEDURE

Using the outputs of References 1 and 2 and other design information, the effect of each failure shall be classified as shown in Paragraph 4 below.

4. METHOD

(1) Table 1 is a list of classification versus condition of the payload after a failure has occurred. (Example - Class 4)

(2) If a failure is time sensitive then it shall be listed as follows:

Class - (#1) - T ([t]

#1 - class until tim. t

#2 - class after time t

t - time (in days) to change class

(Example A 4 becomes a 2 in 2 weeks) Class 4 - T2 (14)

Table 1. Failure Classification
(Condition After Failure Occurs)

Class No.	Remarks
1	Safety hazard (is a safety problem in Shuttle bay; or, if in space, will be if captured by the Shuttle.)
2	Payload lost
3	Payload required ground refurbishment
4	Payload requires ground refurbishment
5	Payload has a SPF leading to 1
6	Payload has a SPF leading to 2
7	Payload has a SPF leading to 3
8	Payload has a SPF leading to 4
9	Experiment failure (25-50% data lost)
10	Experiment failure (10-25% data lost)
11	Experiment failure (< 10% data lost)
12	Major module degradation
13	Minor module degradation
14	Negligible module degradation

APPENDIX N, ATTACHMENT 4
SAMPLE OF FAILURE INDICATOR REQUIREMENT PROCEDURE

1. PURPOSE

The purpose of this Attachment is to recommend requirements of TLM information for various classes of failures.

2. PROCEDURE

The FMEA evaluation personnel shall determine if the Instrumentation and TLM requirement of Table 1 are met for each class of failure. Recommentations shall be made for additional TLM and Instrumentations, with cost tradeoff, as necessary. Also recommendations for reductions should also be made, where applicable.

3. REFERENCE

Table 1 of Attachment 3.

Table 1. Failure Indicator Requirements Chart

Class (See Table 1 of Attachment 3)	Requirement (See code below)
1	A
2	N/A
3	A
4	B
5	A
6	A
7	A
8	C
9	D depends on cost of instrumentation
10	D depends on cost of instrumentation
11	D depends on cost of instrumentation
12	E
13	C
14	E (not required)
15	E (not required)

Code

A - Positive "fail safe" indicator, i.e,, failure indication of indicator failure also required.
B - Positive indicator - Isolated to 1 module
C - Positive indicator - Isolated to 2 modules
D - 50% chance of isolating experiment
E - Indicator, if economically justified